NASA TECHNICAL NOTE

NASA TN D-6226

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NASA TN D-6226

REFERENCE SYSTEM CHARACTERISTICS FOR MANNED STOPOVER MISSIONS TO MARS AND VENUS

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • MARCH 1971

1. Report No. NASA TN D-6226	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle REFERENCE SYSTEM CHARACTERISTICS FOR MANNED STOPOVER MISSIONS TO MARS AND VENUS		5. Report Date March 1971
		6. Performing Organization Code
7. Author(s) Jerry M. Deerwester and Susan M. Norman		8. Performing Organization Report No. A-3665
9. Performing Organization Name and Address NASA Ames Research Center Moffett Field, Calif., 94035		10. Work Unit No. 130-12-06-04-15
		11. Contract or Grant No.
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D. C. 20546		13. Type of Report and Period Covered Technical Note
		14. Sponsoring Agency Code

15. Supplementary Notes

16. Abstract

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The scope of the analysis is as broad as is consistent with a reasonably concise presentation of the results. Missions to both Venus and Mars are included, and, for the latter planet, both orbiting and landing missions are compared. For all the missions, the effects of planetary parking orbit eccentricity and mission duration are shown since each has a strong influence on the total mission requirements. Comparisons are made of various representative high-thrust propulsion system options, including space-storable and cryogenic propellants, nuclear rockets, and aerobraking in appropriate combinations for the various stages.

Analysis results are presented as follows: The MEO requirements for a representative stopover mission to Venus and for stopover missions to Mars during every launch opportunity between 1980-2000 for the most attractive mission profiles are summarized. In addition, the flight times, mission dates, and velocity requirements are given. Finally, the scaling laws used in the MEO calculations are discussed. These scaling laws define the space environment and the spacecraft and propulsion systems. As reported in this document, the results constitute an interim description of present best estimates of future technologies and therefore should provide a useful set of nominal system characteristics and reference missions to serve as a common basis for future studies.

17. Key Words (Suggested by Author(s)) Manned planetary missions		18. Distribution Statement Unclassified — Unlimited	
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 75	22. Price* $3.00

*For sale by the National Technical Information Service, Springfield, Virginia 22151

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SUMMARY

This document sets forth reasonable and consistent trajectories, system elements, features of the natural environment, and overall performance requirement for manned stopover missions to Venus and Mars. Use of the most recent interplanetary trajectory analyses provided a unified framework for comparing mission modes and launch opportunities. The scaling laws that define the spacecraft and propulsion systems and the space environment were established. The trajectory data and the scaling laws were combined and values calculated for the required mass in Earth orbit (MEO) for each of the missions under study.

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INTRODUCTION

Considerable study has been devoted to the definition of the characteristics and requirements of manned planetary missions. This work has ranged from broad comparisons of alternate mission modes, launch opportunities, and spacecraft systems to more detailed examinations of various elements of such missions. Results have been sufficient to achieve significant agreement among the various analysts on many of the mission factors involved.

However, the detailed results of such studies are often at variance regarding the quantitative description of the various environmental characteristics, system elements, and overall performance requirements. This is not to state that some results are "correct" or that others are "wrong." The variations are simply a consequence of the uncertainties inherent in the prediction of future technology.

This document sets forth consistent characteristics for advanced systems and defines the properties of desirable flight profiles for stopover missions to Venus and Mars. This common set of nominal system characteristics and the reference missions can serve as a basis for future studies. None of the details set forth here can be construed as final or "approved" in any sense. Rather, they are provided as an interim description of present best estimates of future technologies. With such reference systems defined and available for baseline purposes, new results that evolve from detailed studies of the system elements will be of more immediate utility.

The results contained in this report were developed as follows: First, use of the most recent interplanetary trajectory analyses provided a unified framework for comparing mission modes and launch opportunities. Second, the scaling laws that define the spacecraft and propulsion systems and the space environment were established from discussions with those who are actively involved in particular technical areas within NASA, results of appropriate contractor studies, and analyses conducted within the Mission Analysis Division. Third, the trajectory data and the scaling laws were combined at the Mission Analysis Division and the required mass in Earth orbit (MEO) calculated for each of the missions under study.

Presentation of the detailed trajectory information and scaling laws is such that the resulting MEO values can be completely reproduced if desired. More important, comparative MEO values can be computed on a consistent basis by any interested user who might wish to investigate the effect of changing a particular input datum or scaling law. (A sample MEO calculation is included in the appendix.) In addition, the MEO results are presented in bar chart form for selected mission modes, launch opportunities, and propulsion system options. These specific results should be of direct use in program planning activities where the major implications of the various alternatives are of interest.

The scope of the analysis is as broad as is consistent with a reasonably concise presentation of the results. Missions to both Venus and Mars have been included for the 1980 to 1999 time period. MEO requirements for Mars vary significantly during this time period. Venus mission requirements, however, are relatively consistent; therefore, the results for missions to Venus are shown for a single typical opportunity. Direct mission profiles to Mars and Venus are examined as well as the Venus swingby mode for Mars missions. In all cases, planetary stopover times of 30 days are considered. The planetary capture mode analyses include both propulsive and aerodynamic capture into circular and elliptic orbits. At Mars, MEO requirements for orbiters and landers are compared, while at Venus only orbiters are analyzed.

The four mission profiles are illustrated in figure 1 for representative flight times. Other flight times are included in the analysis, however, since the flight time has a significant effect on the total mission requirements. During an outbound swingby (fig. 1(a)), the Earth-Mars opposition occurs during the return leg so that the actual launch year is always 1 to 2 years earlier than the year of opposition. For the inbound swingby (fig. 1(b)), however, opposition occurs during the outbound

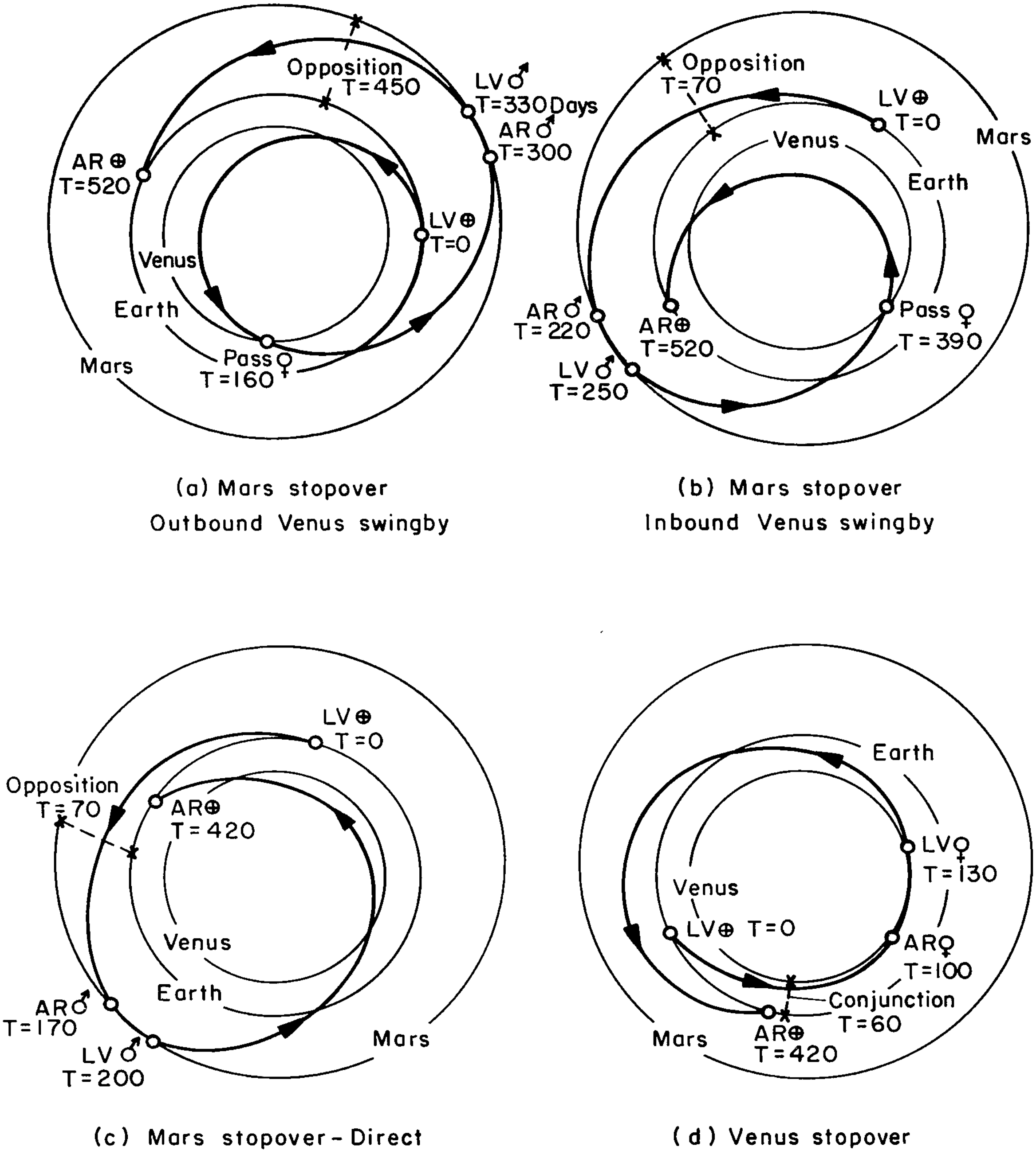


Figure 1.— Representative mission profiles.

leg so that the launch year is usually the same as the year of opposition. Similarly, for the Mars direct mission (fig. 1(c)) and Venus mission (fig. 1(d)) opposition and conjunction, respectively, occur during the outbound leg.

In addition to an examination of the effects of mission mode, capture orbit eccentricity, launch year, and orbiter and lander mass requirements, various representative high-thrust propulsion system options have also been selected for comparison. These include space-storable and cryogenic propellants, nuclear rockets, and aerobraking in appropriate combinations for the various maneuvers. No reusable stages are considered, and aerodynamic reentry is assumed at Earth return; the sensitivity of MEO to such operational philosophies, however, is included.

MASS IN EARTH ORBIT REQUIREMENTS

MEO requirements for manned stopover missions to Venus and Mars are presented in this section. Results are shown for desirable mission profiles and for a wide spectrum of possible propulsion system options. The details of the mission, system, and environmental properties used in the computations are discussed in subsequent sections of this document.

The results presented here are intended primarily to be used on a comparative basis; therefore, the MEO results for any given mission are subject to some qualifications in that certain items have not been included explicitly in the calculations. MEO values will increase above those contained herein because of provisions for launch delays, abort requirements, inclusion of the ascent load bearing shell (discussed under Scaling Laws), assembly in Earth orbit or standardization of modular elements. Quantitative effects of such factors, however, are presented later as appropriate.

The results are displayed in 13 bar charts – one for each propulsion system option that has been considered. Figure 2 is a sample chart that defines the various mission options and legends. On the actual charts, only the mission definitions and the type of propulsion system are labeled explicitly.

For each opposition and mission mode (i.e., direct flight or Venus swingby), eight values of MEO are shown for each propulsion system. At Mars the four values to the left represent the requirements for lander missions, while the four values to the right represent requirements for the orbiter missions. The respective weights off-loaded at Mars are about 55,000 kg (120,000 lb) and 10,000 kg, respectively. For Venus, only one representative mission is shown since the MEO requirements do not change significantly throughout the Earth-Venus synodic cycle. Only orbiter missions are included at Venus. The four left and four right values represent probe weights of 55,000 kg and 10,000 kg, respectively. MEO requirements for other values of off-loading at Mars or Venus can be approximated by linear interpolation between the values shown.

Four MEO values are designated for each of the lander and orbiter missions. The two values on the left apply to circular capture orbits at the planet; those to the right apply to elliptical capture orbits where the eccentricity is held constant at 0.7. MEO requirements for other eccentricities can be accurately estimated by linear interpolation.

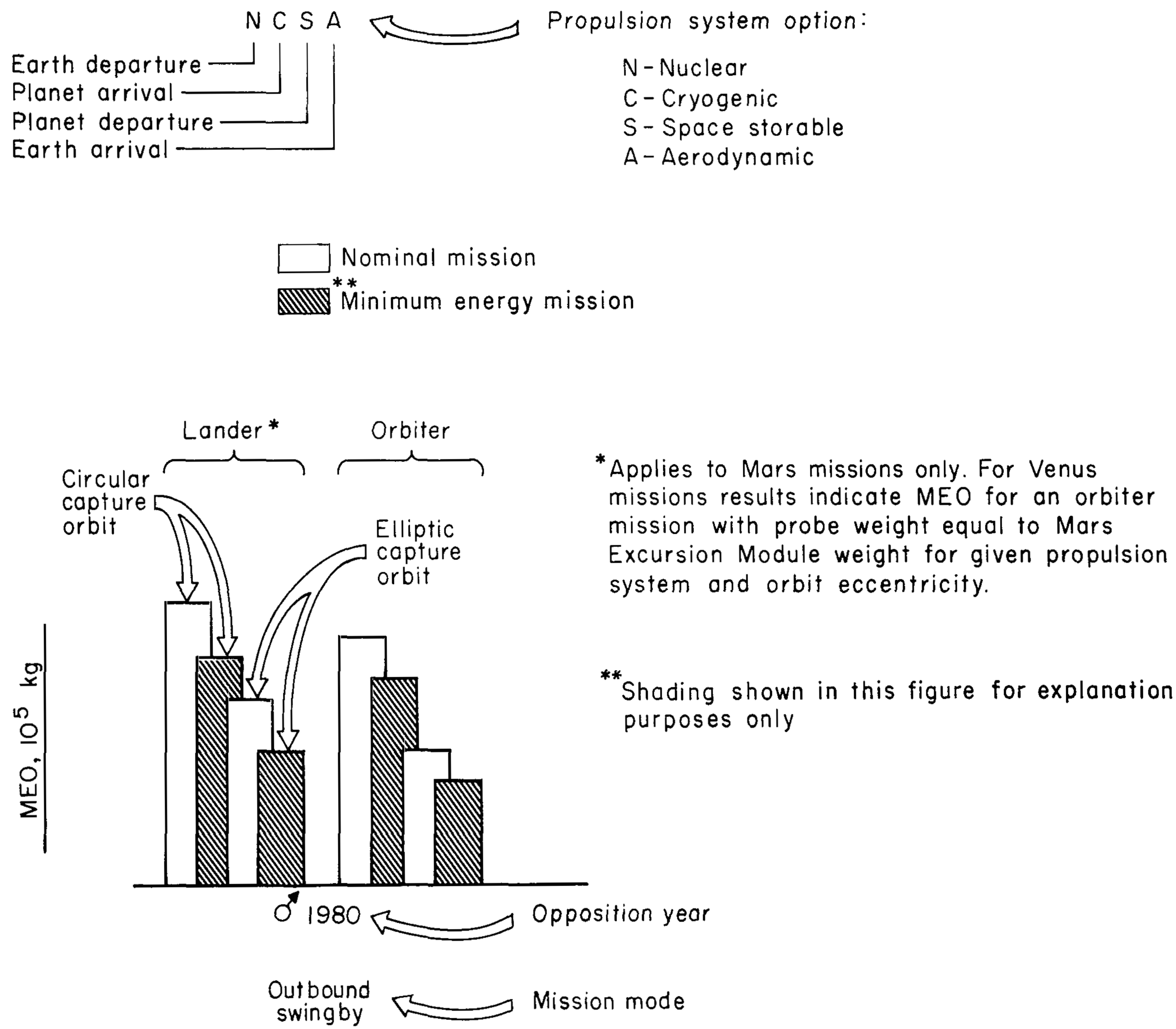


Figure 2.– Mass in Earth orbit requirements – explanatory chart.

The left-hand MEO value for each type of parking orbit refers to a "nominal" mission; the value to the right refers to a minimum energy mission. These two criteria for selection of the mission are explained in the next section. The differences in MEO between such missions are due to variations in mission duration. Typically, for the direct flight mode to Mars or Venus, the mission durations are 400 days and 460 days for the nominal and minimum energy missions, respectively; for the Venus swingby mode, corresponding typical values are 480 days and 620 days.

At the top of each chart, the spacecraft propulsion system applicable to that chart is designated by a combination of the letters A, C, N, and S, each of which refers to one of the following systems:

*A*erodynamic deceleration. This mode is an optional mode of capture at the planet; a vehicle with an L/D of 1.0 is assumed. Also because the Earth entry speeds associated with the more desirable mission profiles are usually moderate, this mode is used at Earth entry without prior propulsive deceleration.

*C*ryogenic propulsion systems. This analysis considers O_2/H_2 systems developing a specific impulse of 450 sec.

*N*uclear propulsion systems. A specific impulse of 850 sec is taken as a representative value.

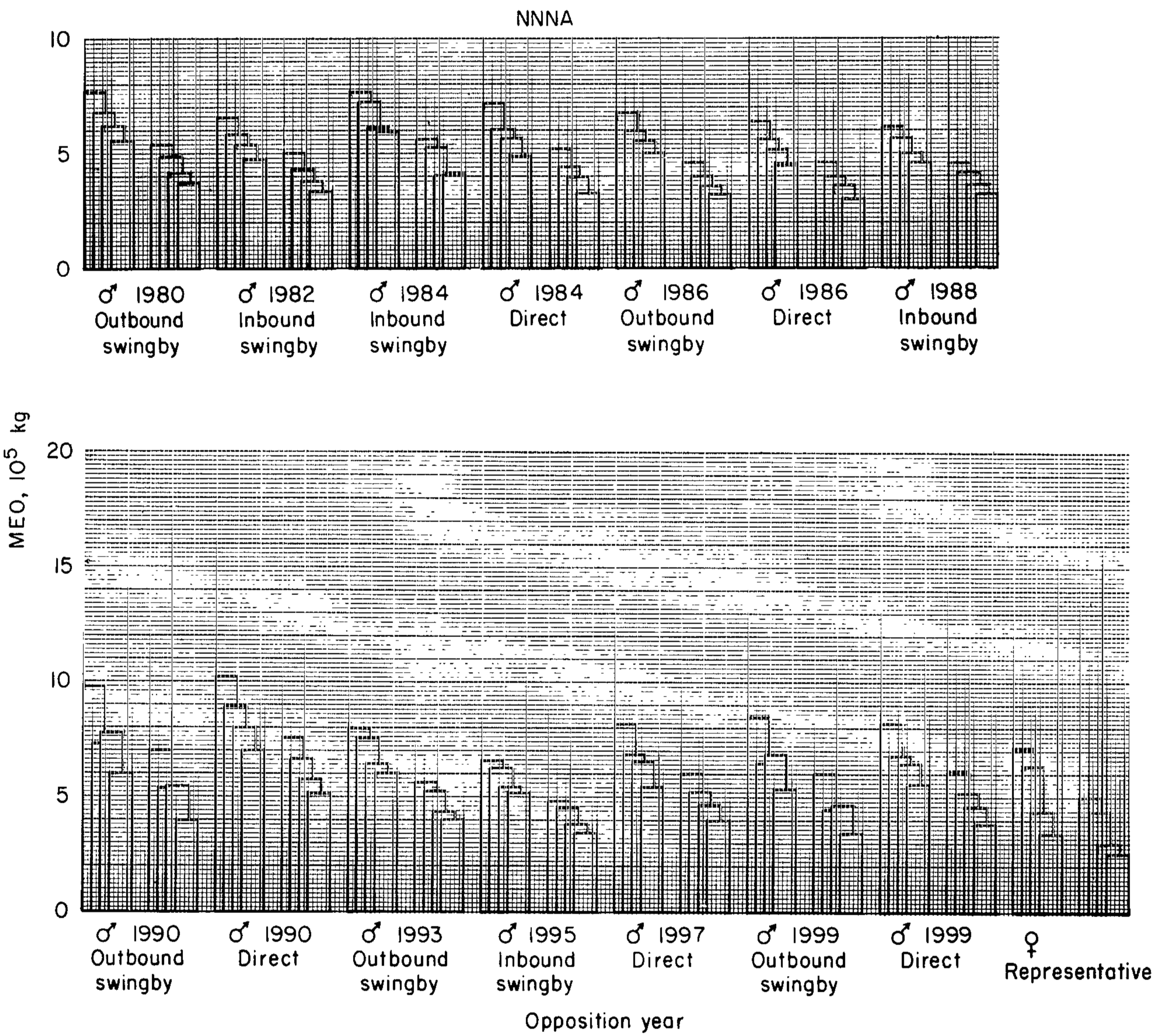
*S*pace-storable propulsion system class. A representative system, $FLOX/CH_4$, is considered here, and develops a specific impulse of 405 sec.

The characteristics of these propulsion systems are discussed in more detail under Scaling Laws. In each four letter sequence, the leftmost letter indicates the system used at Earth departure. Continuing toward the right, each letter represents the type of propulsion system employed at planet arrival, planet departure, and Earth arrival, respectively. All propulsion system combinations that seem even marginally desirable are considered; combinations that are obviously inappropriate are not included. For example, "CSNA" is not given since the development and/or use of three entirely different propulsion systems is not likely – and certainly not in the sequence indicated.

MEO requirements were determined for every combination of mission mode, propulsion system option, and opposition year. Each bar chart, starting in the upper left corner, shows the mission profile that yields a discernible minimum in the MEO requirements for the 1980 Earth-Mars opposition. Reading toward the right, the most attractive profile for each subsequent opposition is indicated. Note that during all but the 1997 opposition, the Venus swingby mode is included.[1] During those mission opportunities in which neither the direct flight mode nor the Venus swingby mode is substantially superior from an MEO standpoint, results for both modes are shown.

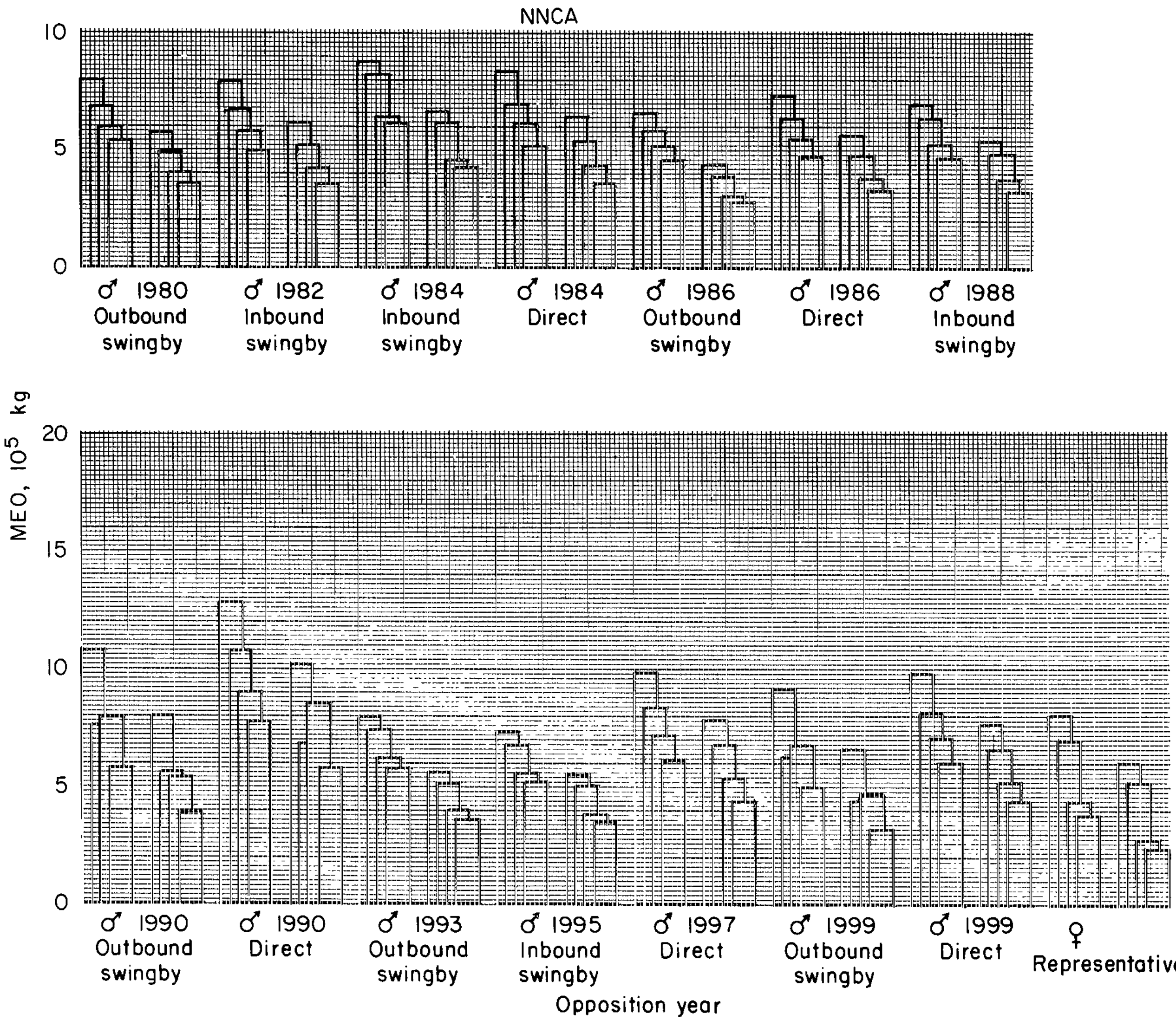
The numerical results are shown in figures 3(a) – (g) and 4(a) – (f) for propulsive and atmospheric capture at the planet, respectively. The MEO requirements are shown in units of 10^5 kg. (This unit is nearly equivalent to the MEO capability of the Saturn V.) The system characteristics employed in the analysis reflect estimated technology circa 1975 and thus, strictly speaking, apply only to flight operations in the 1980's. These same assumed characteristics, however, were also applied to the mission opportunities in the 1990's.

[1] As discussed in the next section, direct trips during five opposition periods were eliminated because of excessive velocity requirements.



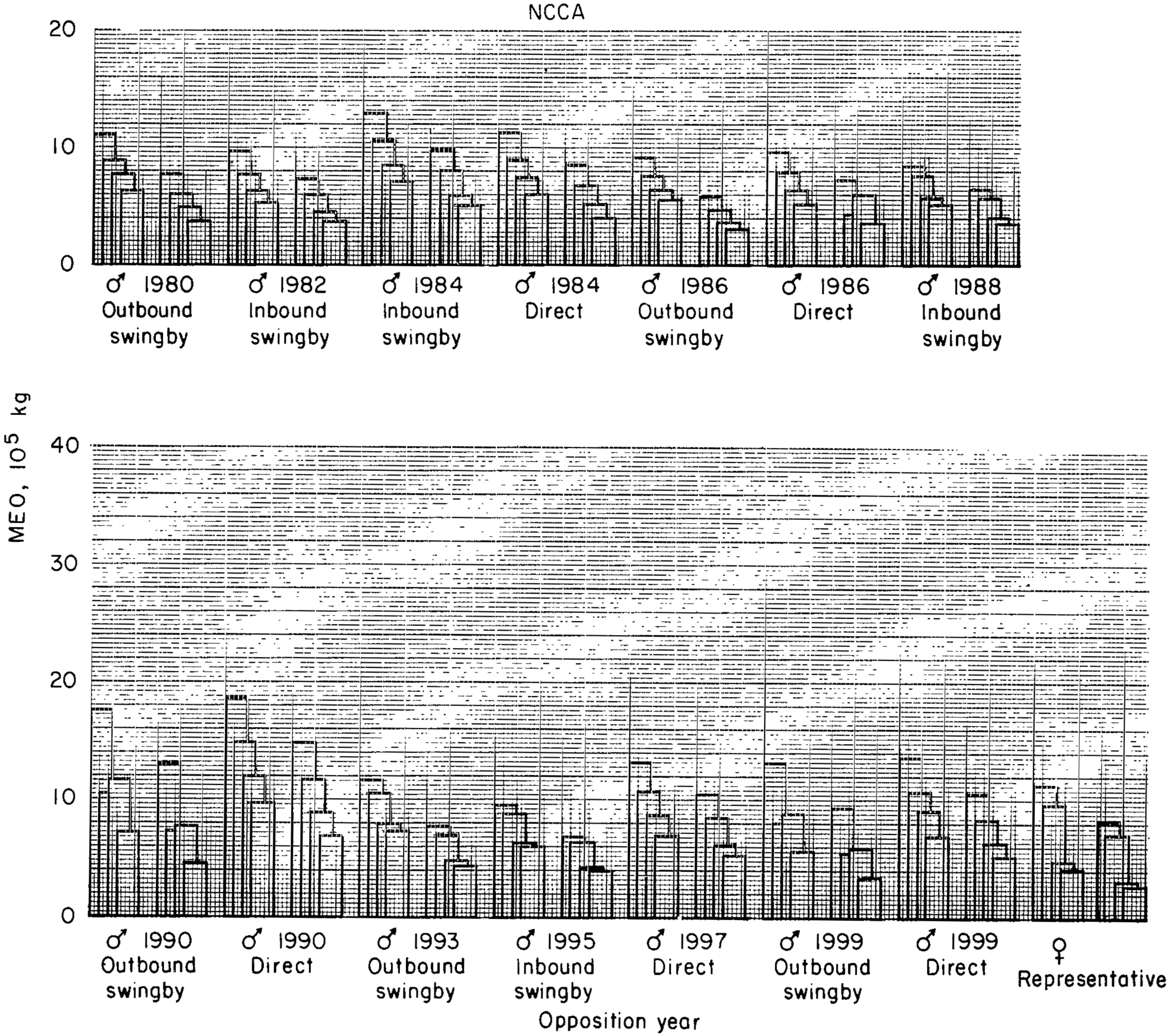
(a) NNNA

Figure 3.– Mass in Earth orbit requirements.



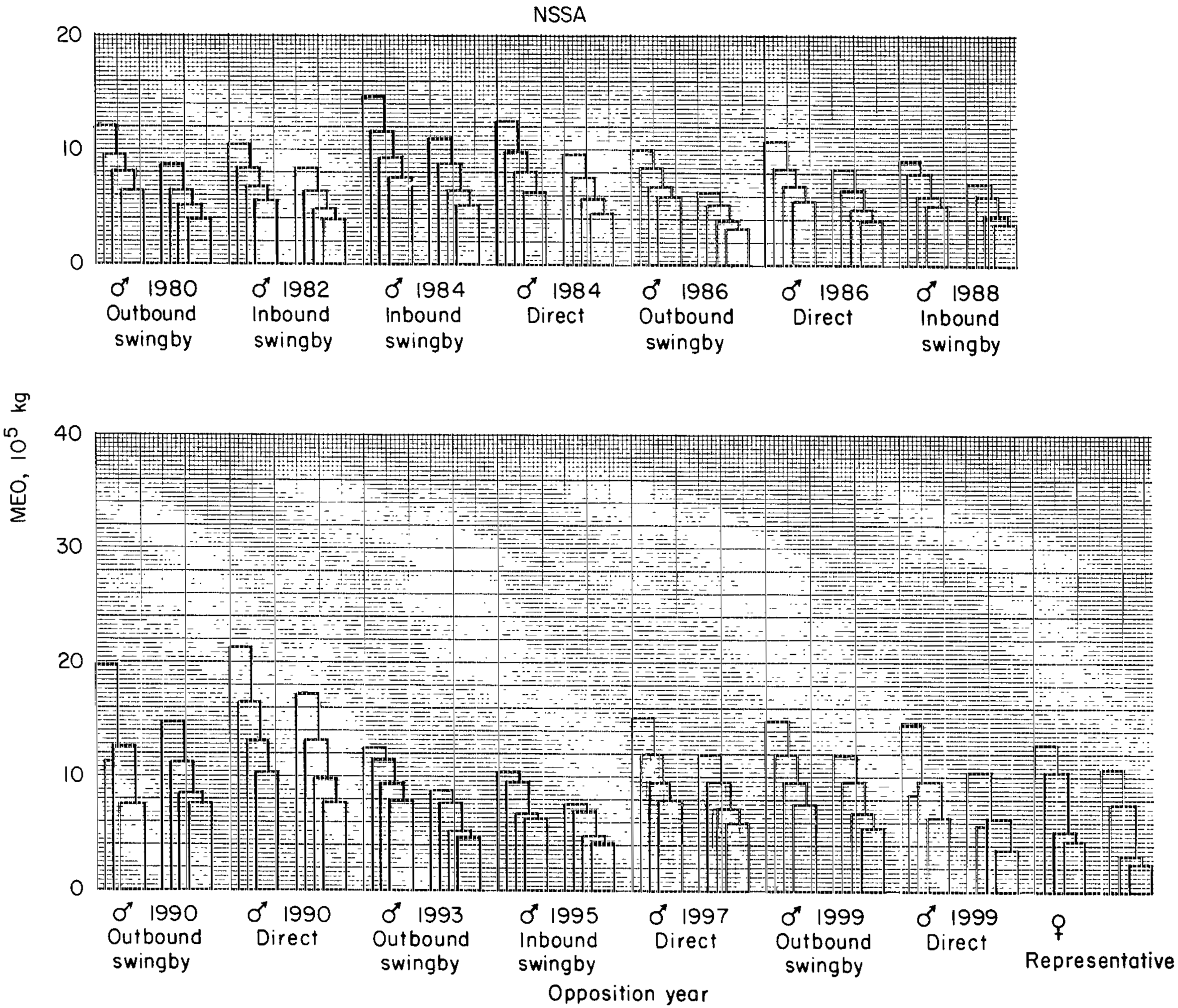
(b) NNCA

Figure 3.– Continued.



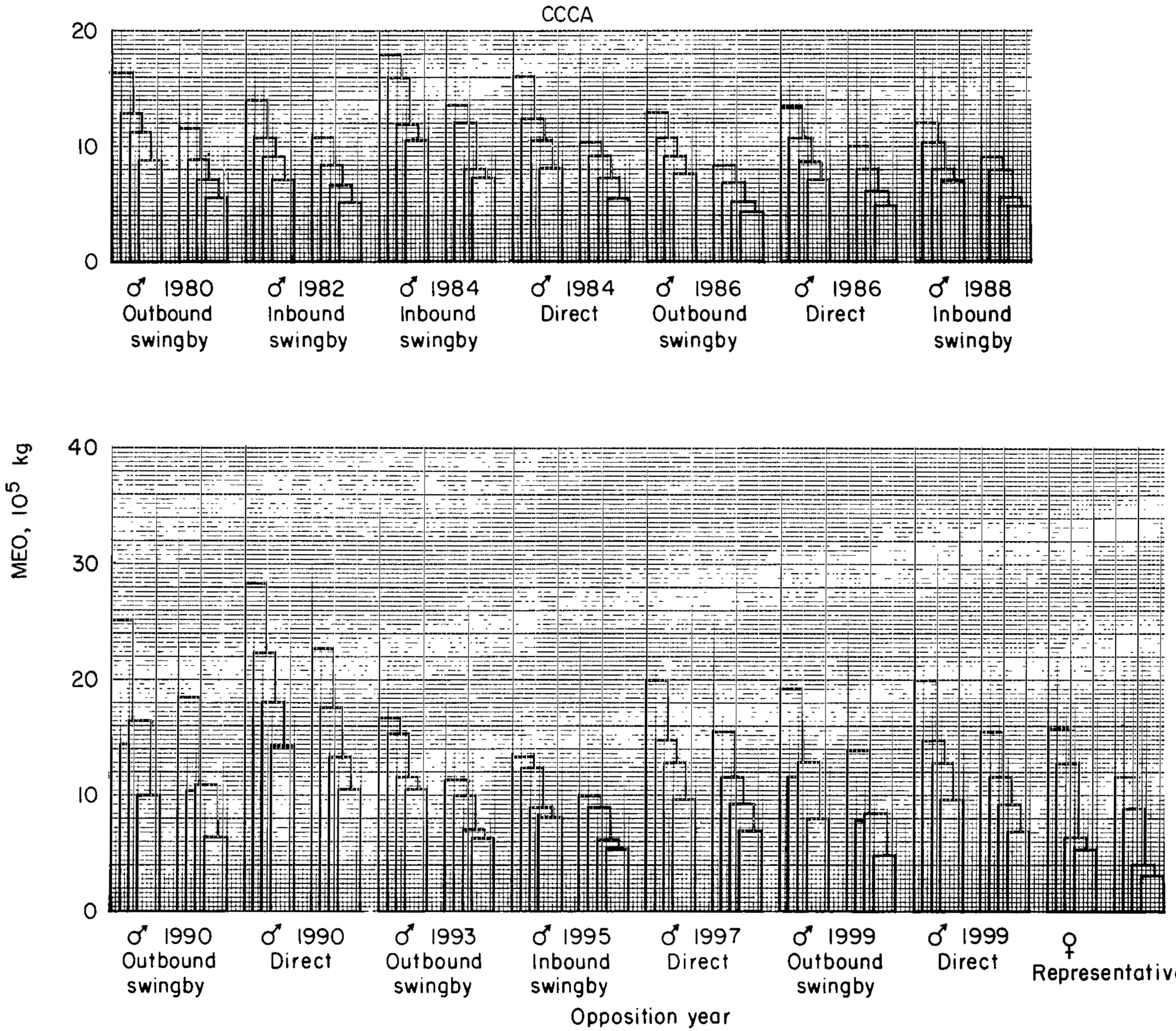
(c) NCCA

Figure 3.– Continued.



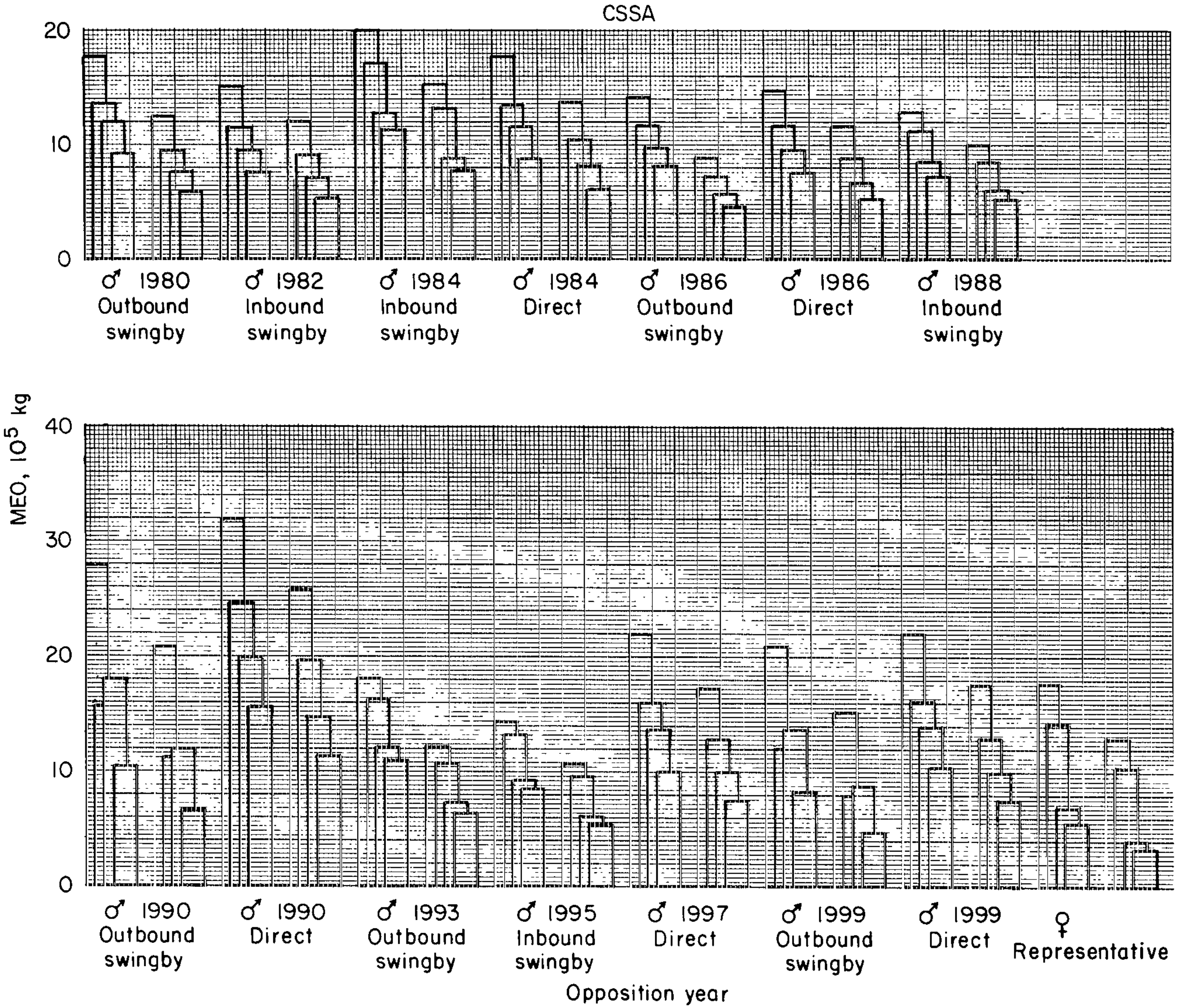
(d) NSSA

Figure 3.– Continued.



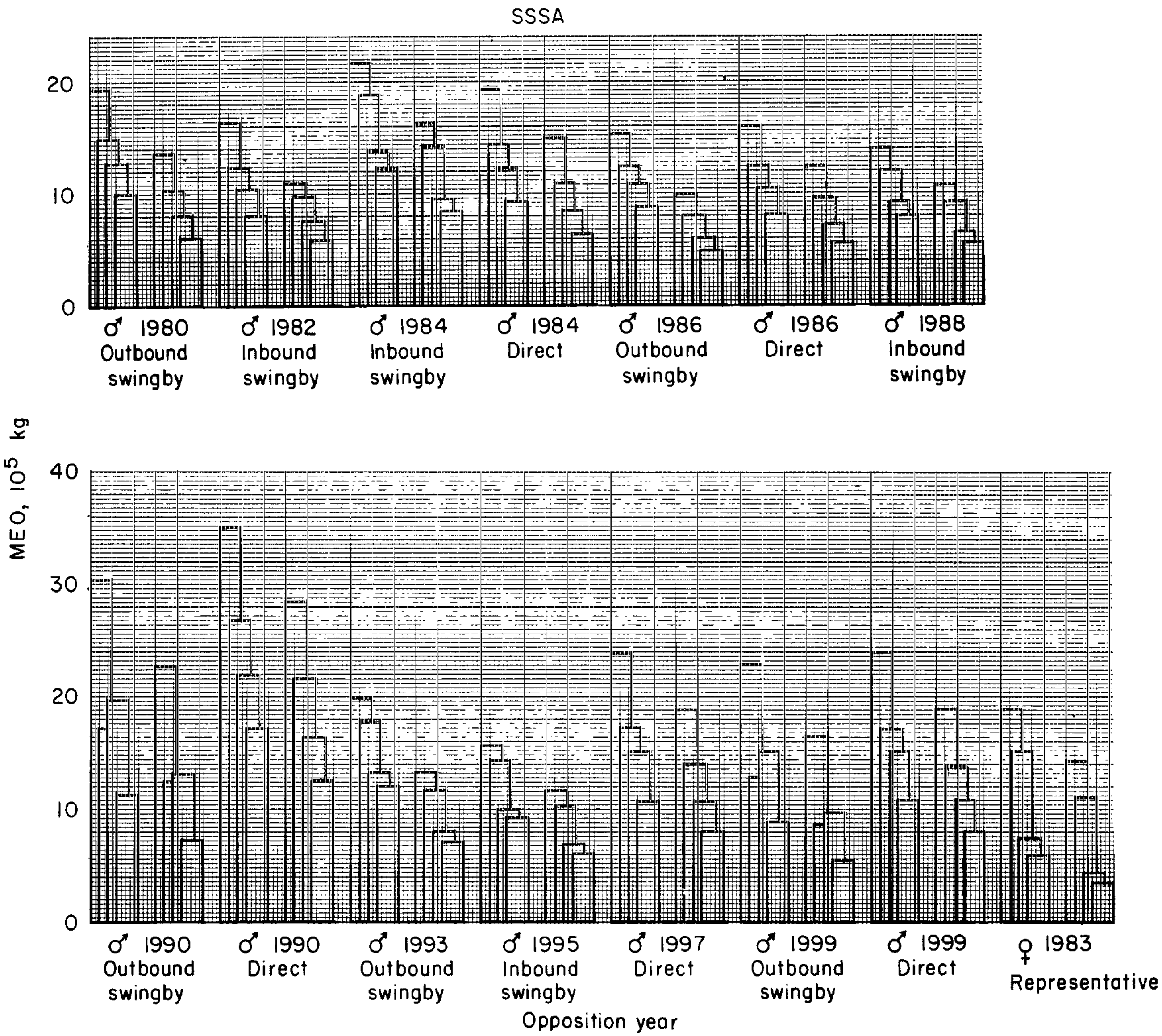
(e) CCCA

Figure 3.– Continued.



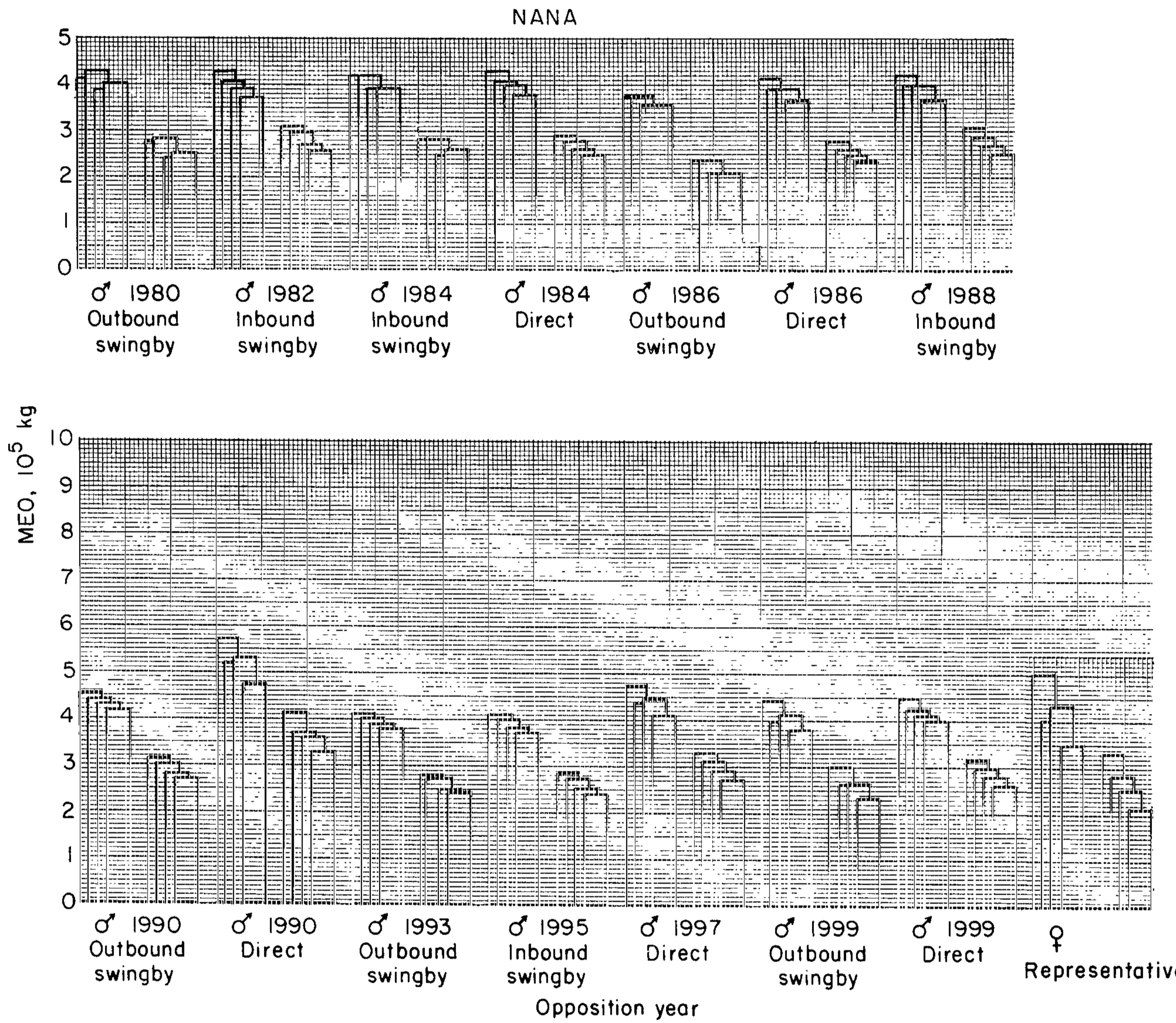
(f) CSSA

Figure 3.– Continued.



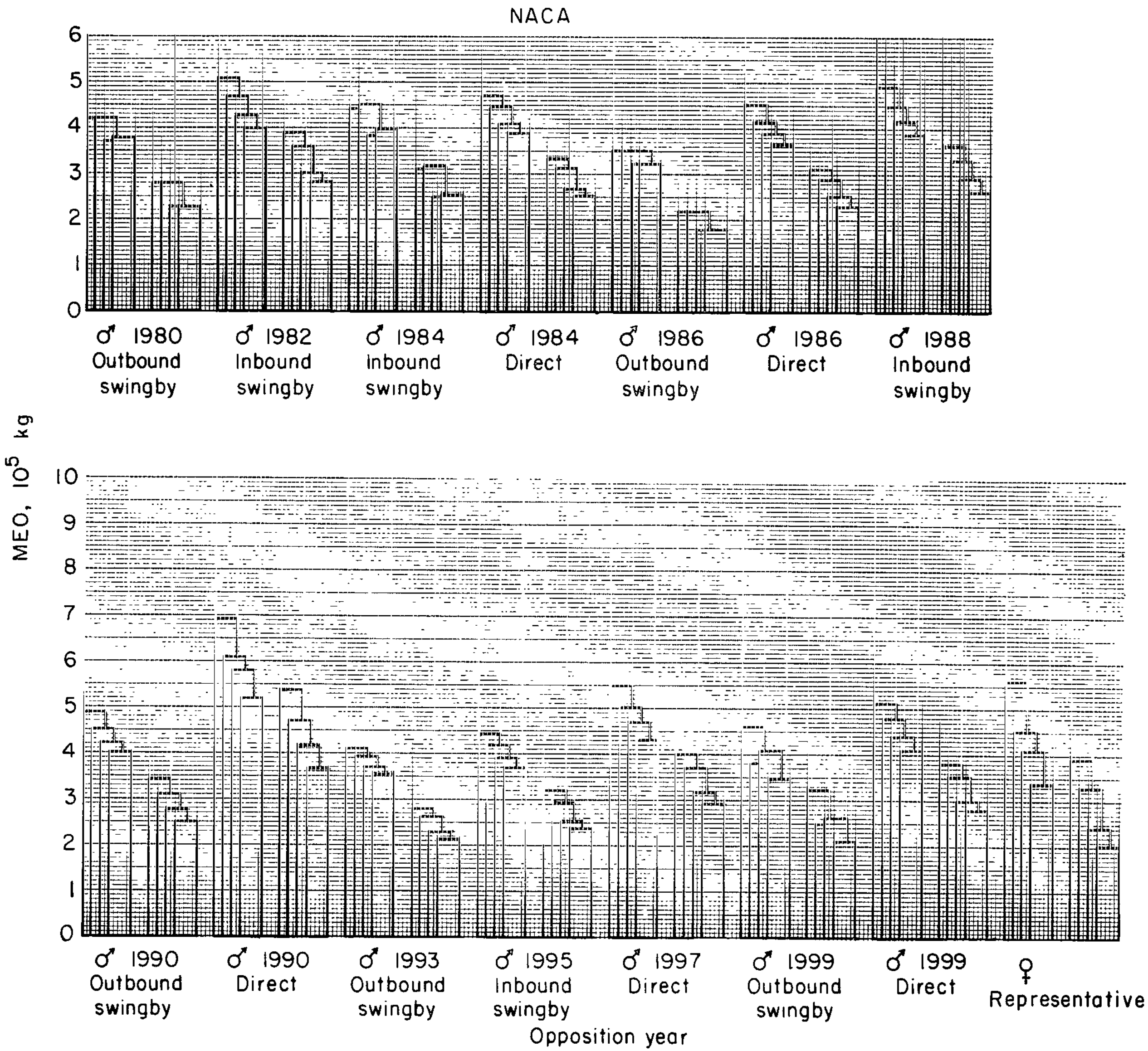
(g) SSSA

Figure 3.– Concluded.



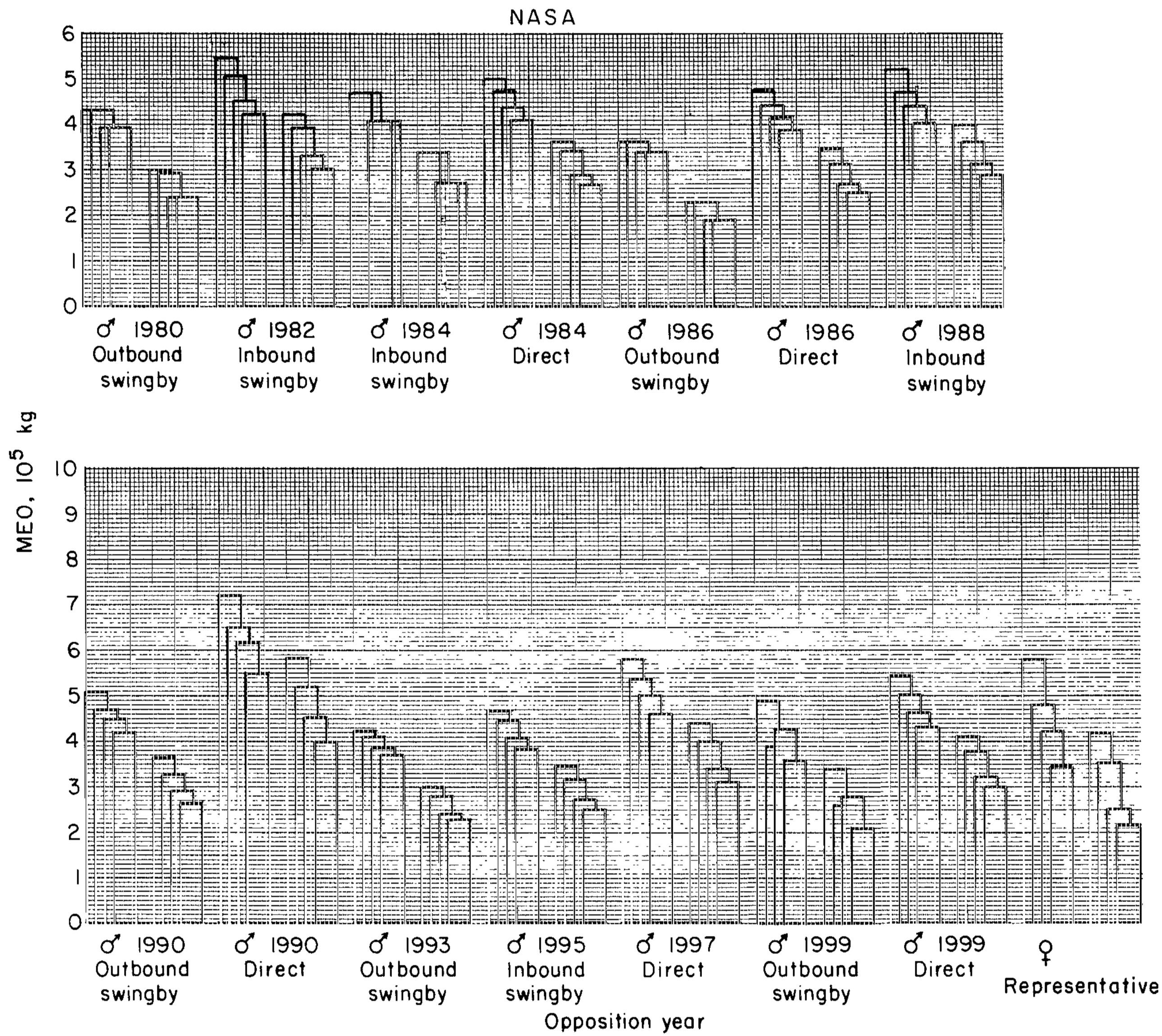
(a) NANA

Figure 4.– Mass in Earth orbit requirements.



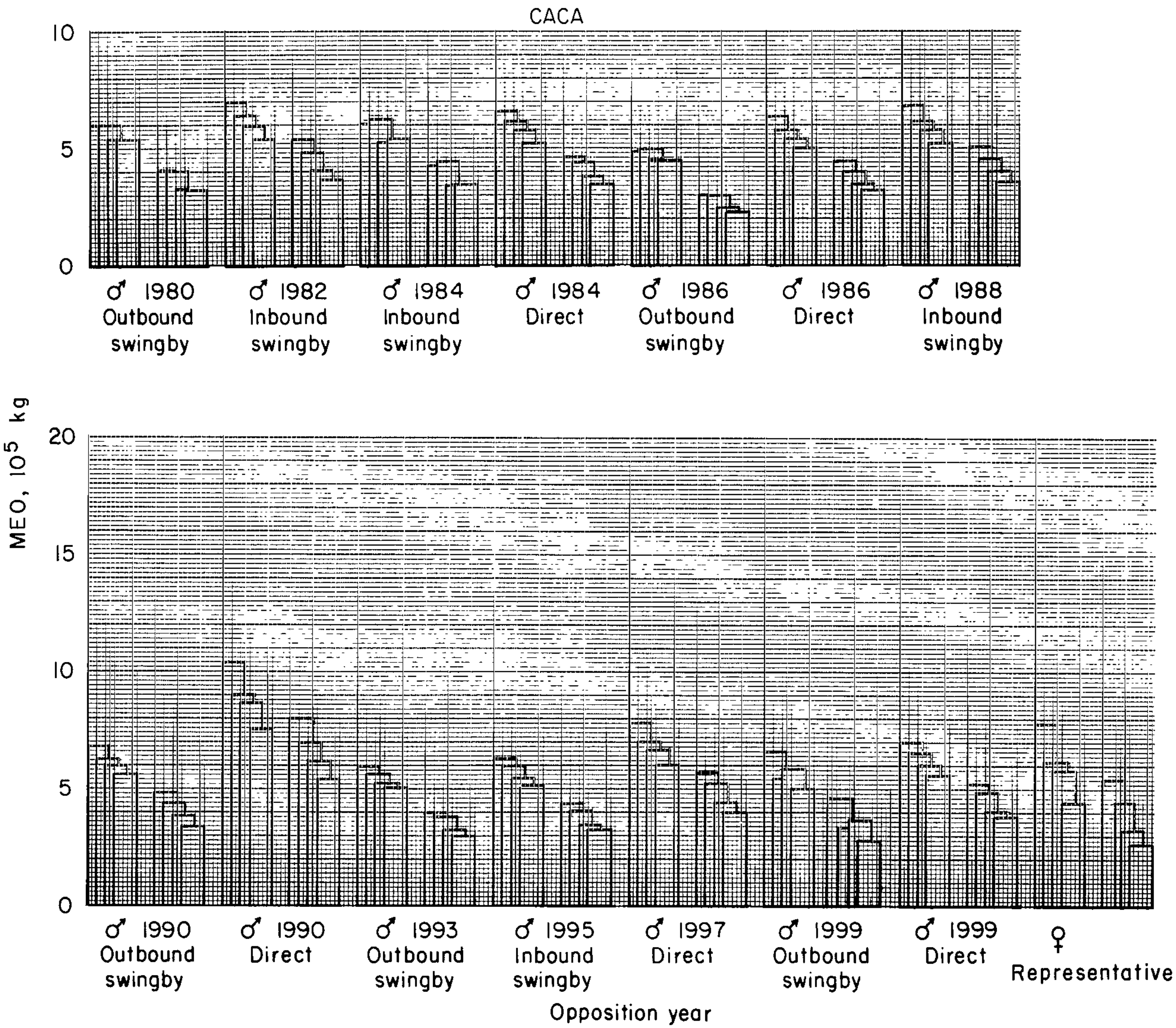
(b) NACA

Figure 4.– Continued.



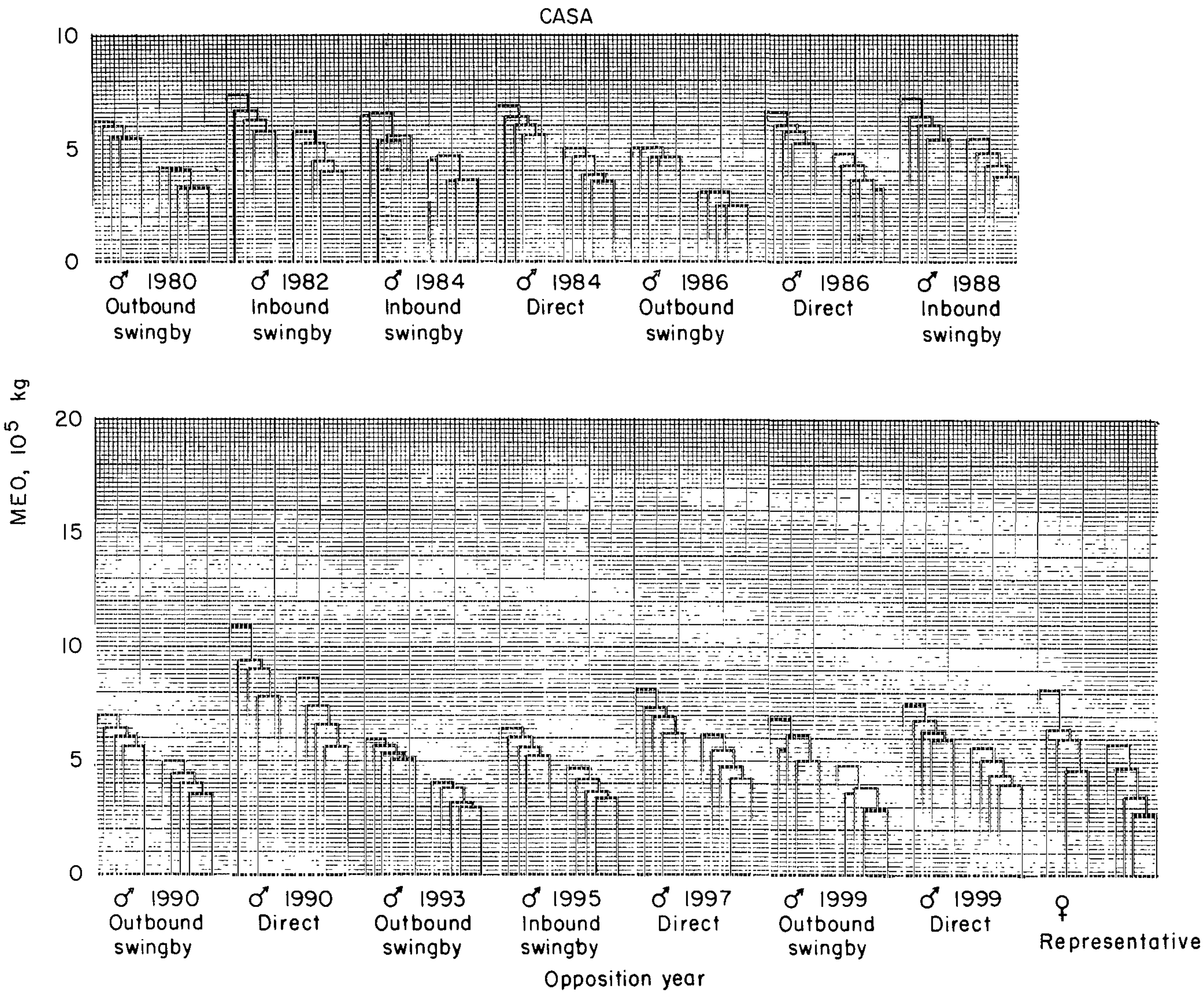
(c) NASA

Figure 4.– Continued.



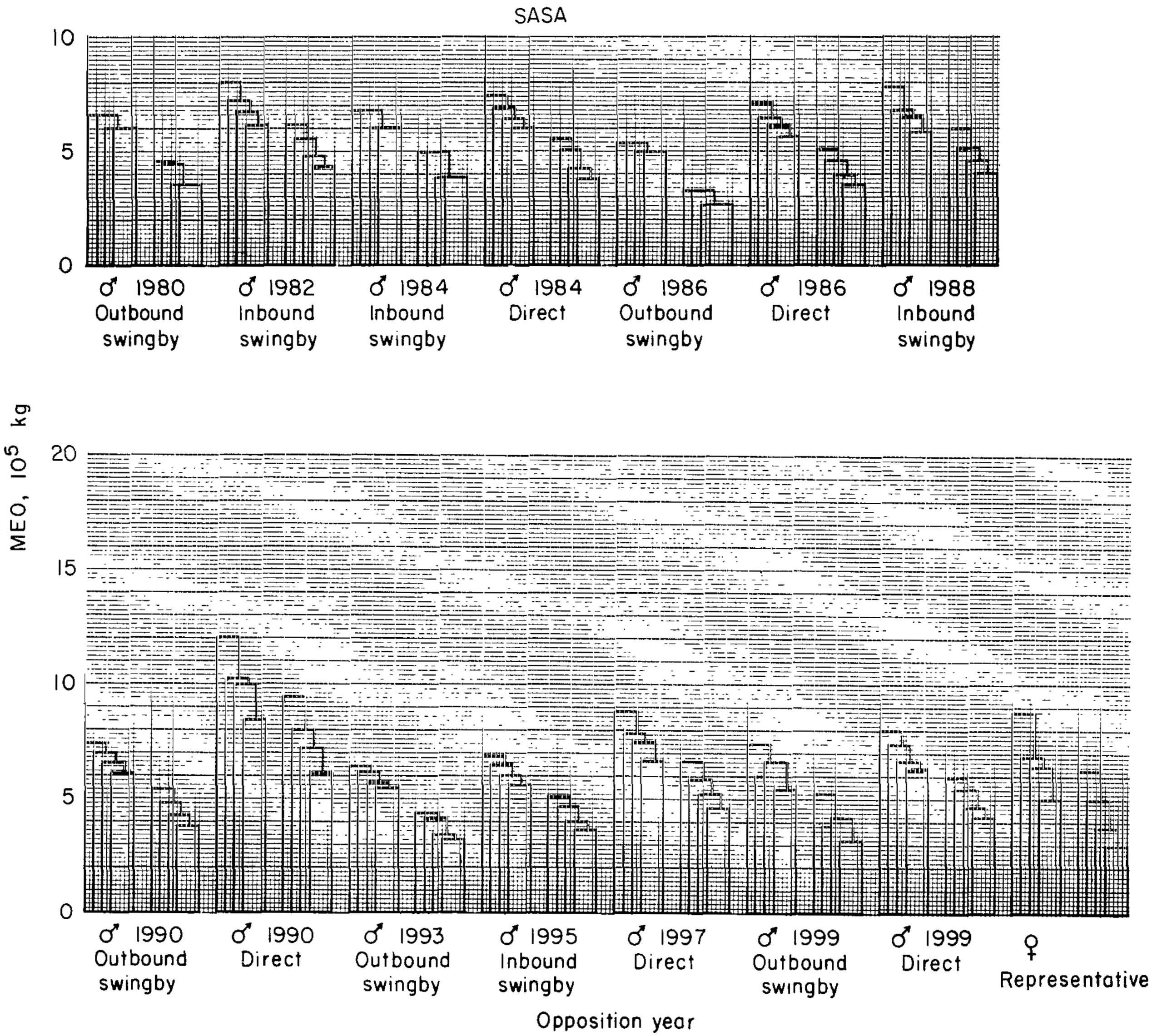
(d) CACA

Figure 4.– Continued.



(e) CASA

Figure 4.– Continued.



(f) SASA

Figure 4.– Concluded.

For each capture mode, the figure sequence is organized generally in order of increasing MEO requirements. Obviously, there are sizable variations in these requirements when propulsion systems and capture modes of such diverse characteristics are considered. Of significance to program planning activities, however, is the equally obvious conclusion that regardless of the propulsion system(s) that may be developed, missions to both Venus and Mars can be scheduled during any launch opportunity. A several-fold variation in MEO requirements throughout the Mars synodic cycle need no longer be considered. Variations that do exist in the requirements could be further reduced by eliminating from consideration missions in the few remaining abnormal opportunities (e.g., 1984, 1990, and 1997).

MISSION DESCRIPTION

This section presents the salient trajectory characteristics of the direct flight mode and the Venus swingby mode of round trip stopover missions to Mars. The results are based on the data of references 1 and 2. In many cases, these characteristics lead to a specification of the most desirable flight modes during a given opposition year without recourse to subsequent MEO calculations. The modes are listed in terms of the year of Earth-Mars opposition rather than launch year. Only one representative stopover mission to Venus is included since the velocity requirements are very similar for other conjunction years (ref. 1). The effects of capture orbit eccentricity and launch delay penalties on the velocity requirements are also briefly discussed.

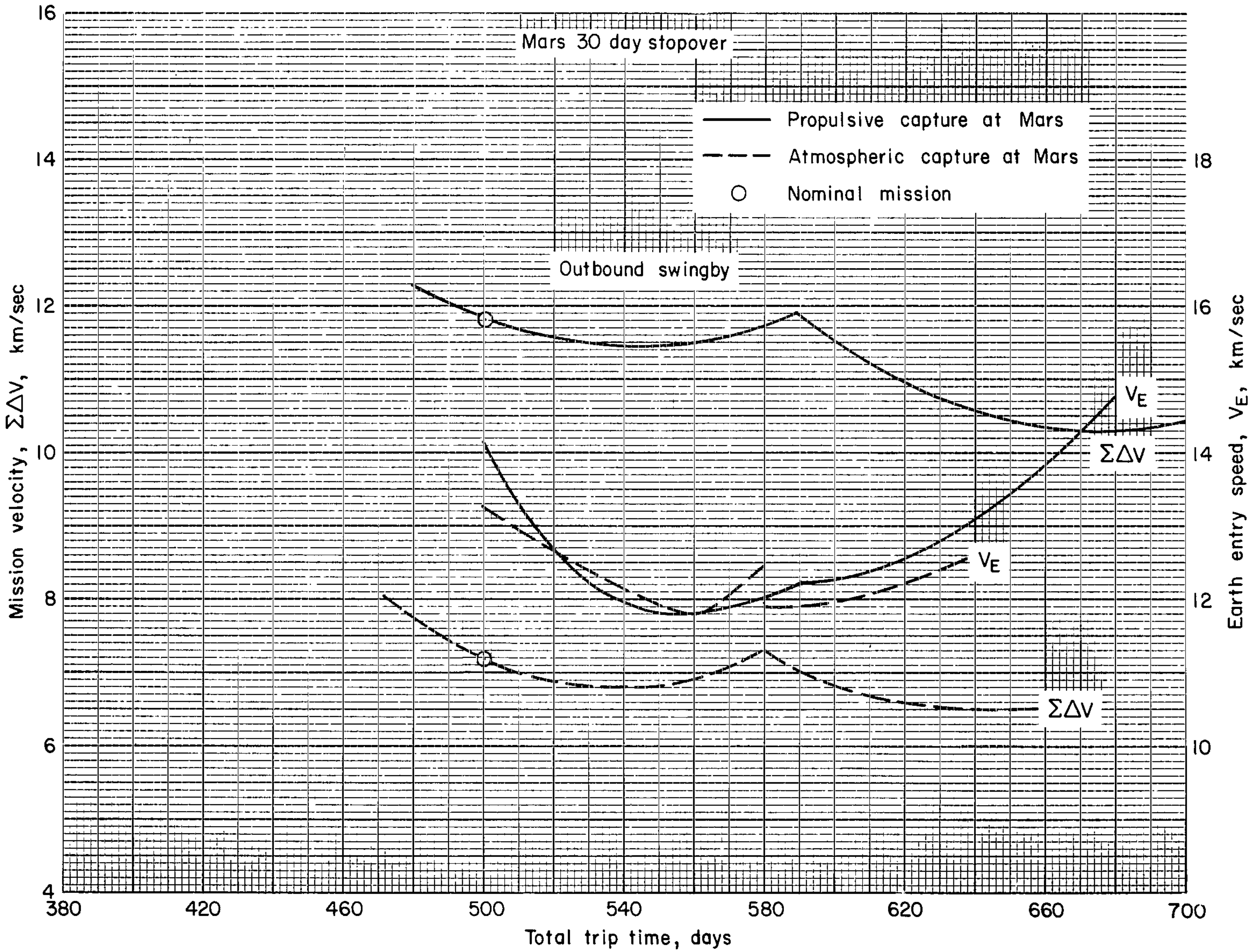
Trajectory Characteristics

Based on the mission velocity requirements, the flight modes listed below should be eliminated from further consideration in program planning activities (ref. 3):

1980 direct
1982 direct
1988 direct
1993 direct
1995 direct
1997 Venus swingby

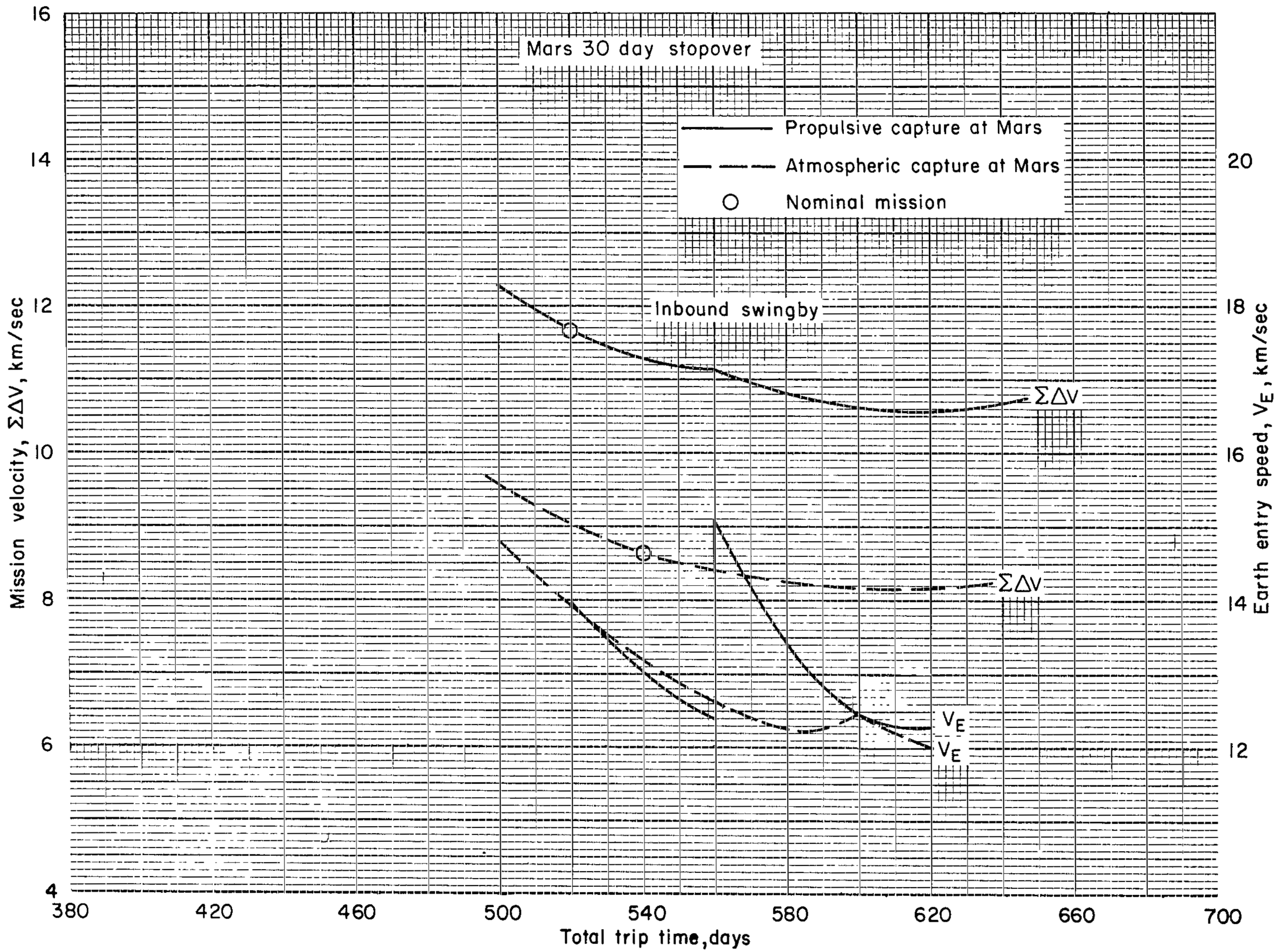
Mission velocity requirements ($\Sigma\Delta V$) and Earth entry speeds (V_e) associated with the remaining missions are shown in figures 5(a) – (k) as a function of mission duration for propulsive and atmospheric capture into a circular orbit at the planet. The data apply to stopover times of 30 days. Data for stopover times of 0 and 60 days are given in reference 2. These data indicate that the changes in mission velocity requirements are generally not more than 10 percent so long as mission durations are allowed to change. The mission velocity consists of the Earth and planet departure impulsive velocity increments for atmospheric capture and includes the planet arrival impulsive velocity increment for propulsive capture. Each curve was developed by selecting departure and arrival dates at Earth and the planet that minimize $\Sigma\Delta V$ for each trip time on the curve.[2]

[2] Alternatively, MEO might be the criteria used in the selection of these dates. Evidence indicates (e.g., refs. 3, 4), however, that this additional complexity is unnecessary as it usually results in departure and arrival dates that are identical with those determined by the procedure used here.



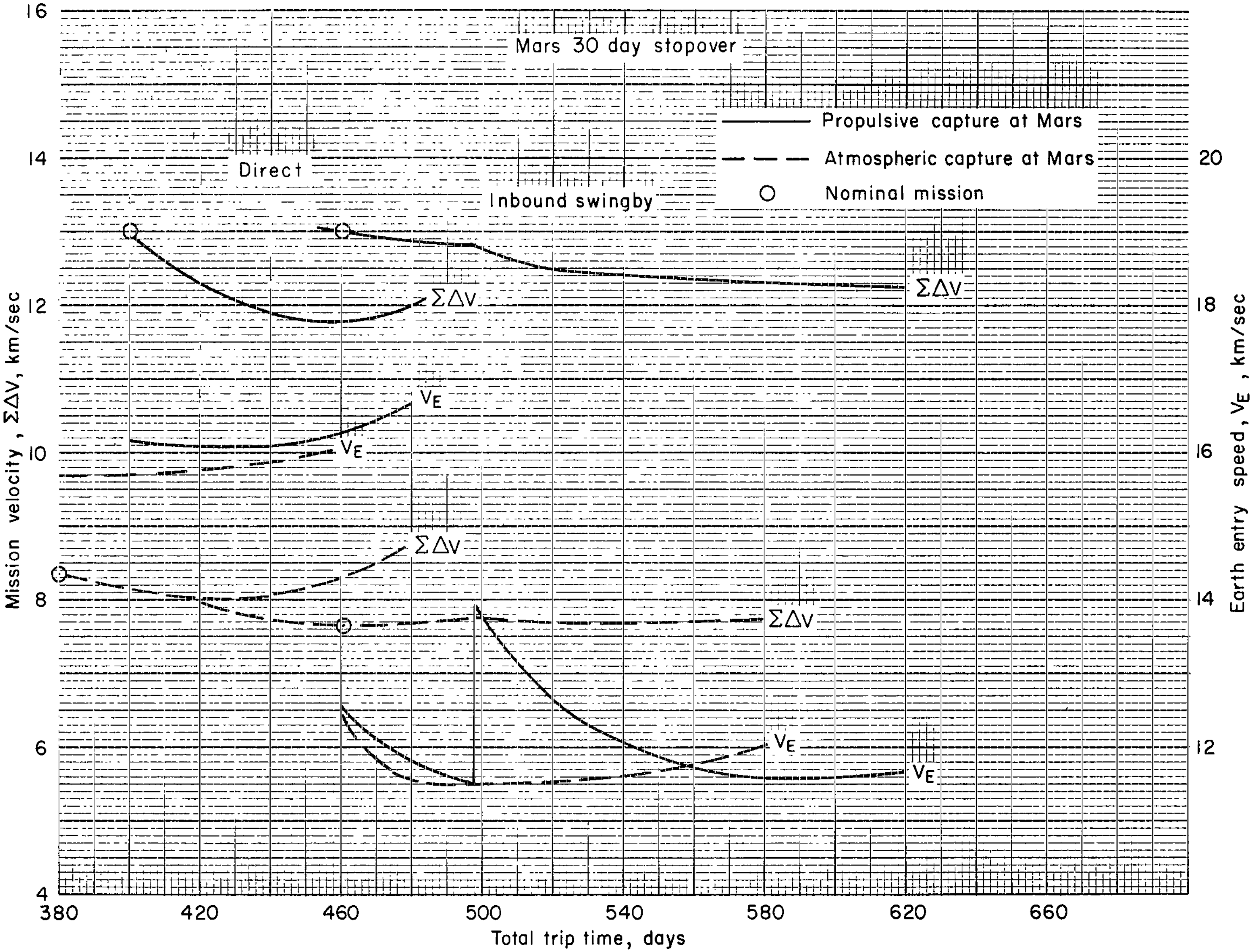
(a) 1980 opposition.

Figure 5.– Mission velocity and Earth entry speed vs. total trip time.



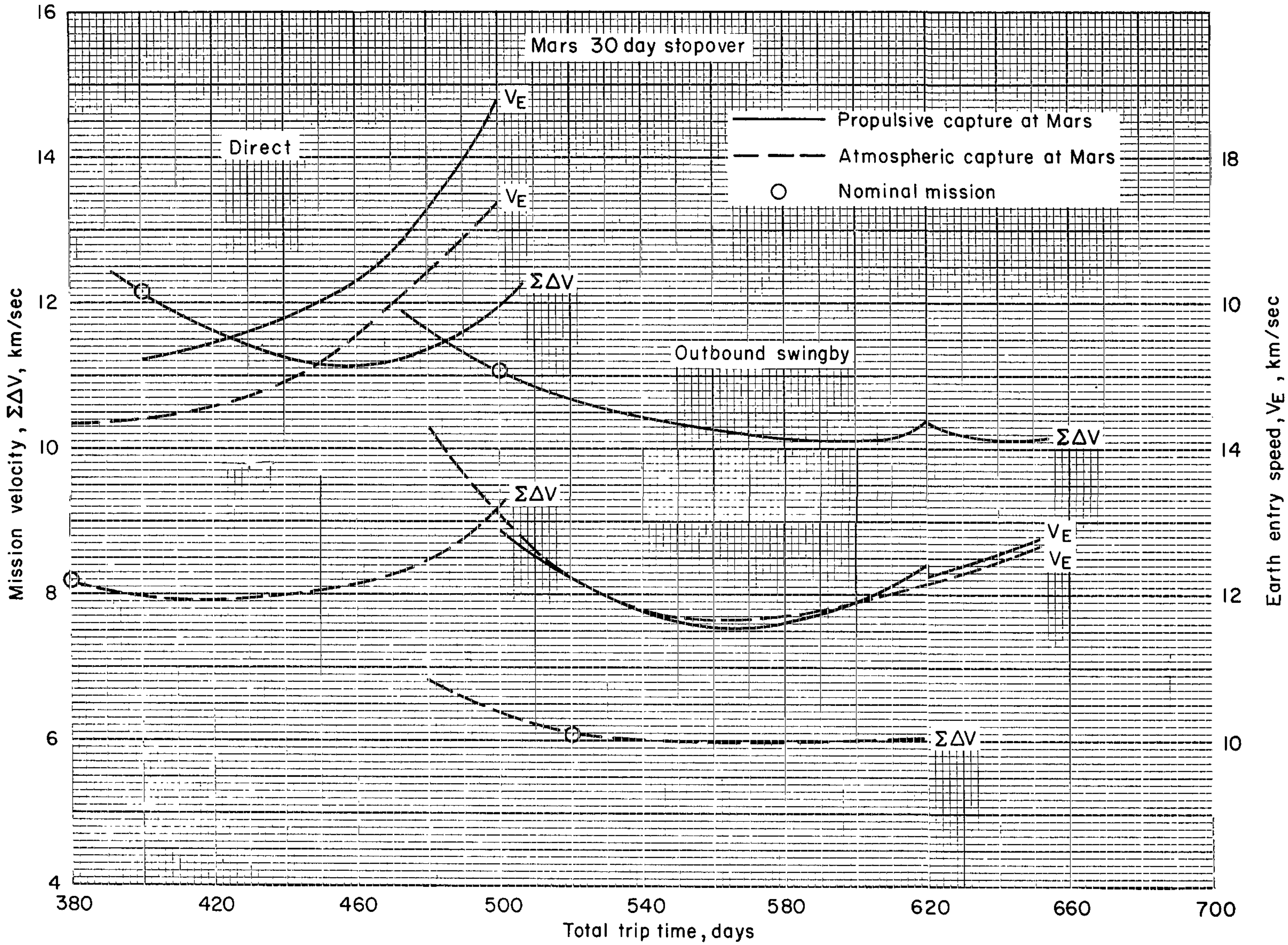
(b) 1982 opposition.

Figure 5.– Continued.



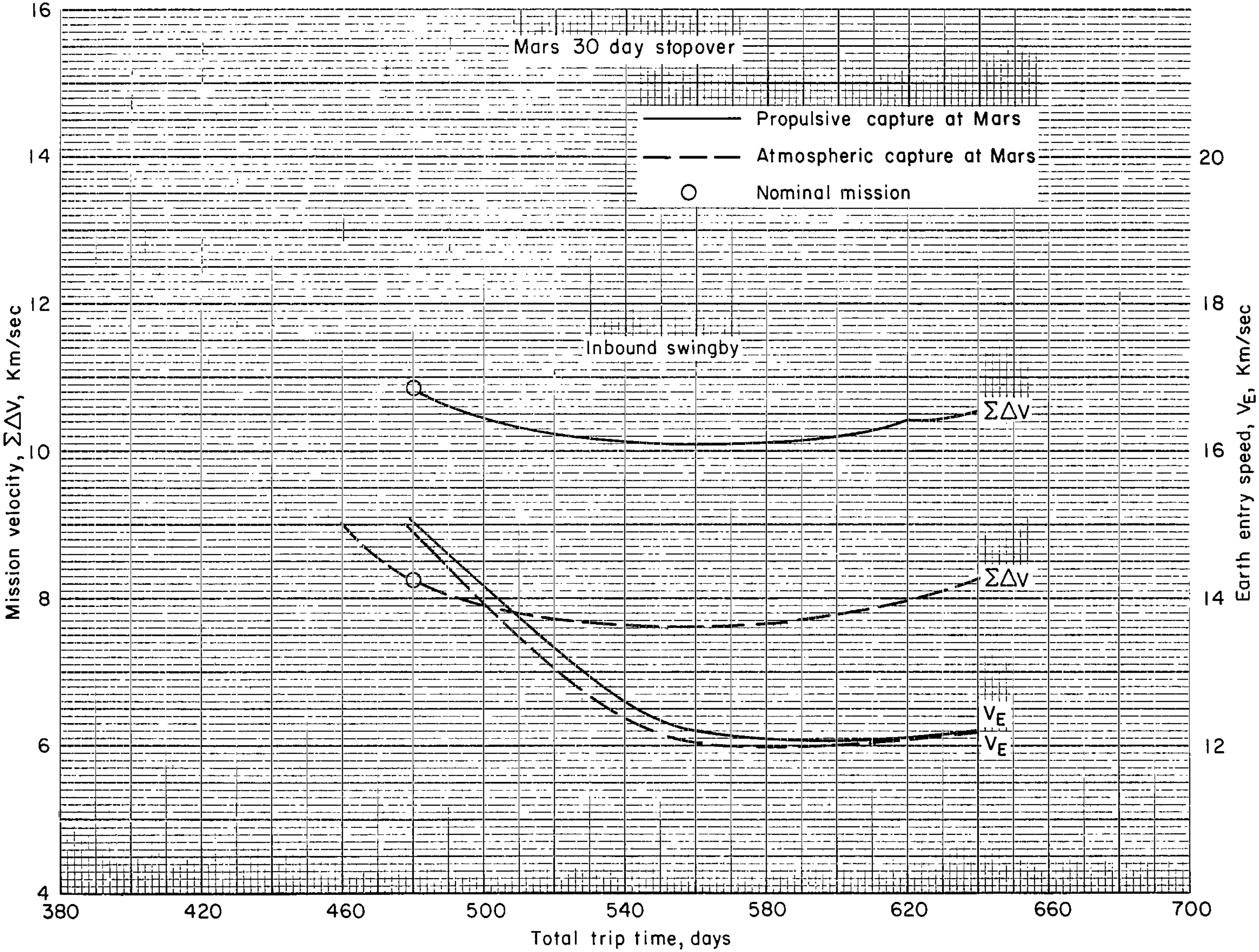
(c) 1984 opposition.

Figure 5.– Continued.



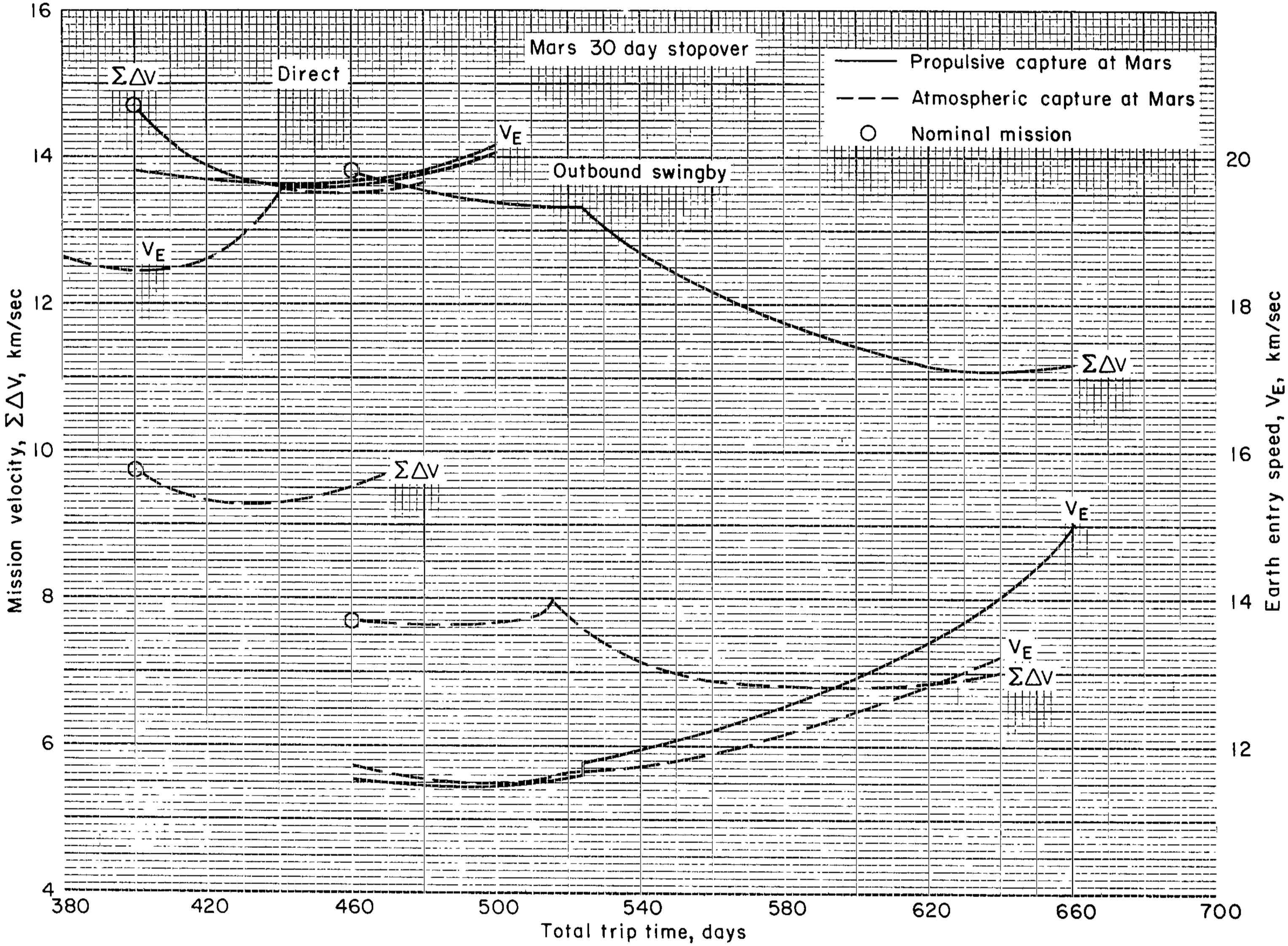
(d) 1986 opposition.

Figure 5.– Continued.



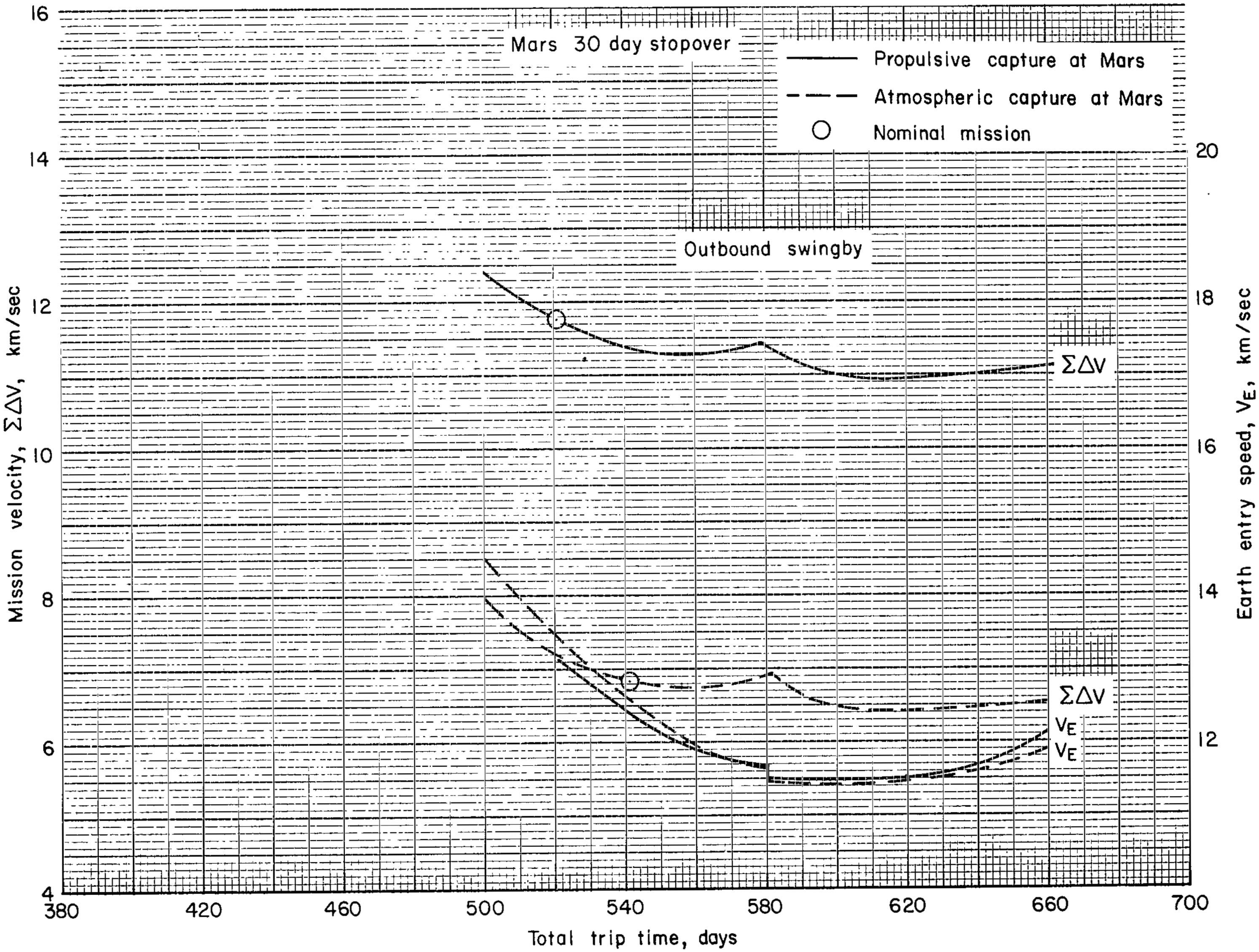
(e) 1988 opposition.

Figure 5.– Continued.



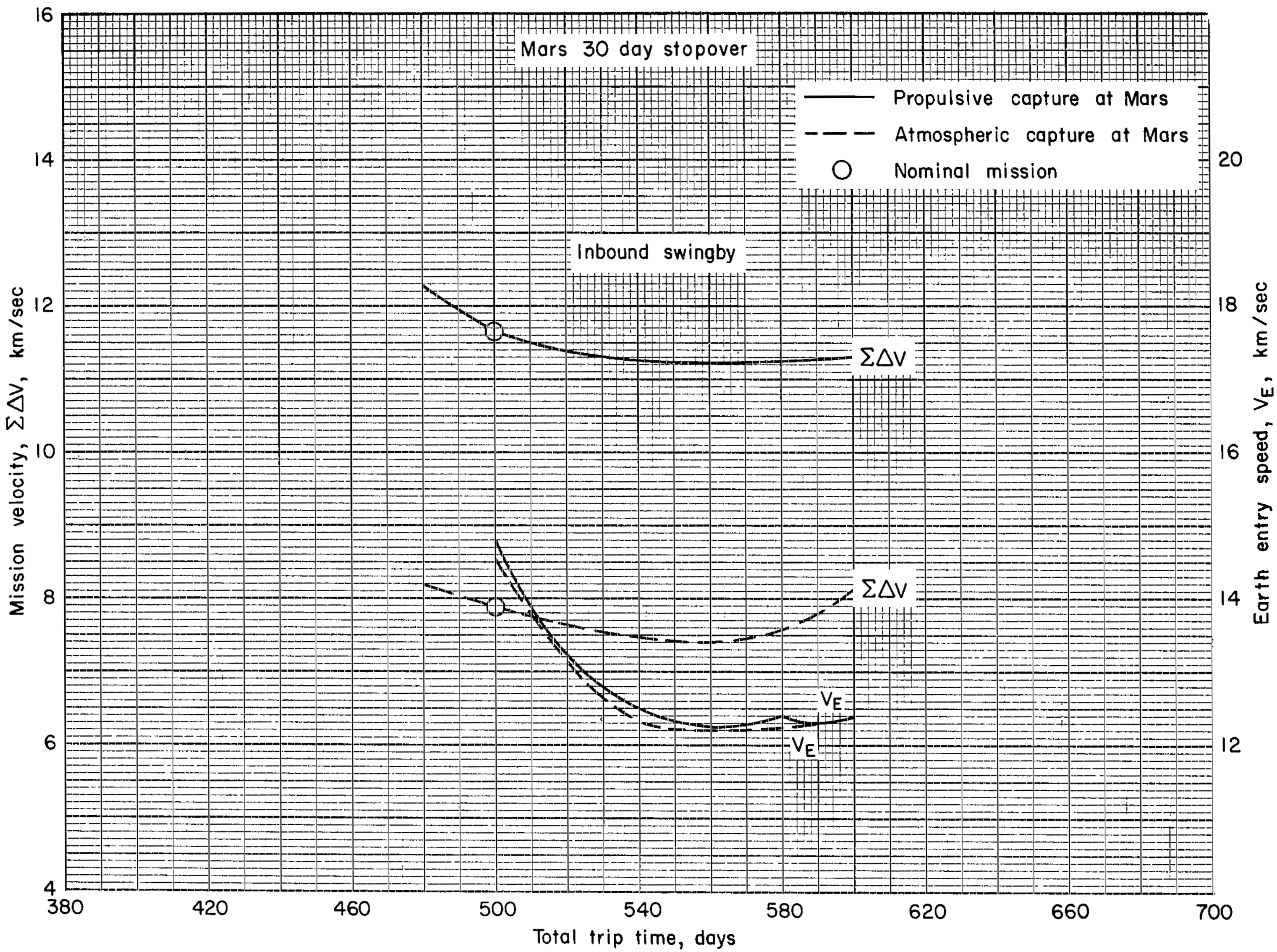
(f) 1990 opposition.

Figure 5.– Continued.



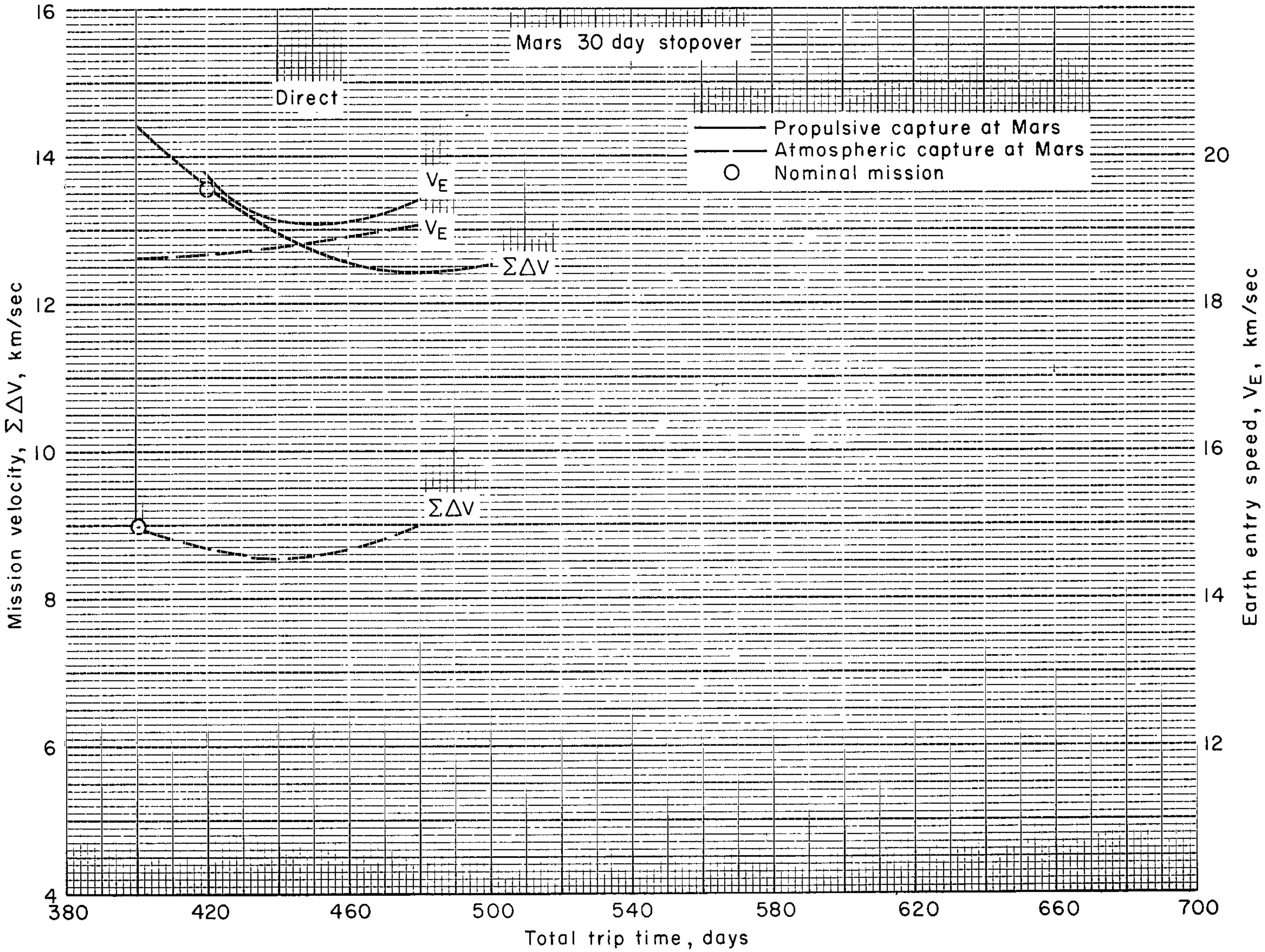
(g) 1993 opposition.

Figure 5.– Continued.



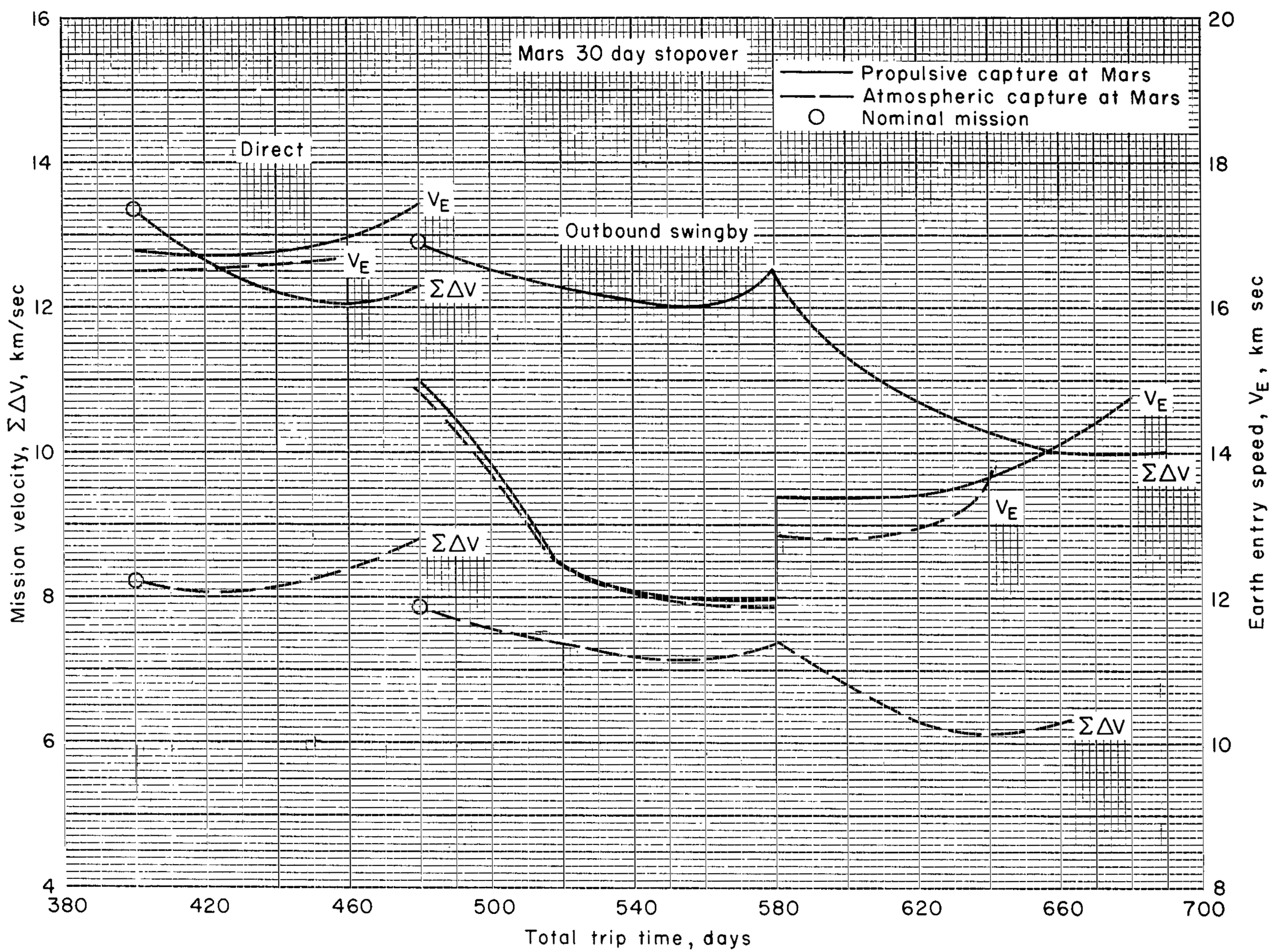
(h) 1995 opposition.

Figure 5. – Continued.



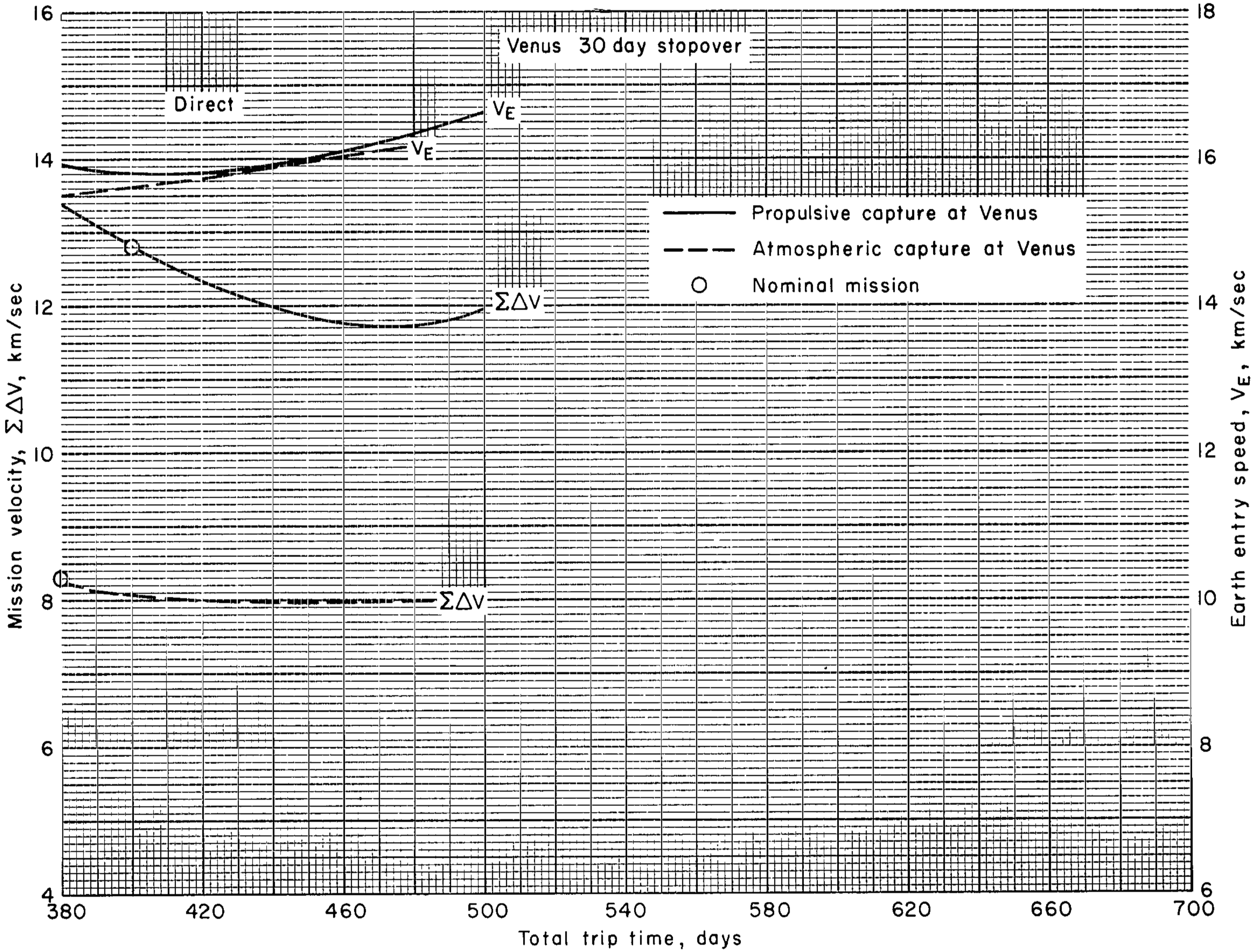
(i) 1997 opposition.

Figure 5.– Continued.



(j) 1999 opposition.

Figure 5.– Continued.



(k) 1983 conjunction.

Figure 5.– Concluded.

Given the trip time-$\Sigma\Delta V$ curves, it is desirable to select one or two trajectories to be used in the MEO comparison. A common mission selection criterion is one of minimizing MEO. Although the weights of various system elements increase with trip time (e.g., life support system, meteoroid protection, cryogenic propellant insulation), such a mission can usually be approximated by minimizing $\Sigma\Delta V$. Two factors of major importance, however, remain unaccounted for in a minimum MEO mission, namely, system reliability and psychological and physiological effects of long duration space missions on the crew. While it is not possible to assess these factors quantitatively at present, it is recognized that mission durations shorter than those for which MEO is minimized may well be desirable. Thus, a second criterion for the selection of missions that may satisfy these qualitative requirements, and can easily and consistently be applied to trajectory data alone, is that of simply minimizing the product of total trip time and mission velocity requirements. These missions, indicated as "nominal" missions in figure 5 result in total trip times approximately 25 percent shorter than those of minimum mission velocity (or minimum MEO) at the expense of less than 10-percent increase in total mission velocity. The effects of these two distinct missions on MEO requirements are shown in figures 3 and 4.[3]

Figure 5 indicates that while the direct and swingby modes can be favorably compared on the basis of $\Sigma\Delta V$, a similar comparison on the basis of V_e may not be favorable. In the 1984 opposition, for example, the direct mode velocity requirements are much less than those for the swingby mode. Consequently, the MEO is also less. As shown in figure 5(c), however, the Earth reentry speeds associated with the direct mode are much higher. Regardless of the year of opposition the direct mode entry speeds are never below about 15 km/sec (49,000 ft/sec) and are usually much higher, whereas the swingby mode reentry speeds are never above about 14 km/sec. For swingbys where the nominal mission may exceed this limit, an increase in the trip time of about 20 days will reduce the entry speed to not more than this value. Even greater reductions in V_e to a range of 11.5 to 12.5 km/sec (i.e., less than 41,000 ft/sec) can be achieved for all of the swingby missions with trip times of no more than 555 days.

Since low entry speeds and short trip times are desirable, in some opposition years both flight modes are included even though one flight mode is obviously superior to the other in terms of velocity requirements. In particular, in view of the desirability of low entry speeds, MEO data were computed in some opposition years for the swingby mode even though the mission velocity requirements may be somewhat higher than those associated with the direct missions. Conversely, because the flight times are generally shorter for the direct mode, MEO requirements were computed for both modes in some years even though the velocity requirements are somewhat higher for the direct mode.

All the energy requirements are expressed in terms of impulsive velocity increments. Since hyperbolic excess speed frequently is used instead, velocity increments required to enter or leave circular orbits as functions of hyperbolic excess speeds at Earth, Venus, and Mars are provided in figure 6.

The detailed characteristics of the selected missions are listed in tables 1(a) and 1(b) for propulsive and atmospheric capture, respectively. All the pertinent information about the missions,

[3] A few cases may be noted in figure 4 for which the MEO is less for the nominal mission than for the minimum energy mission. These cases occur when the savings in propellant requirements afforded by the longer mission durations are counteracted by the attendant increase in mission module weight.

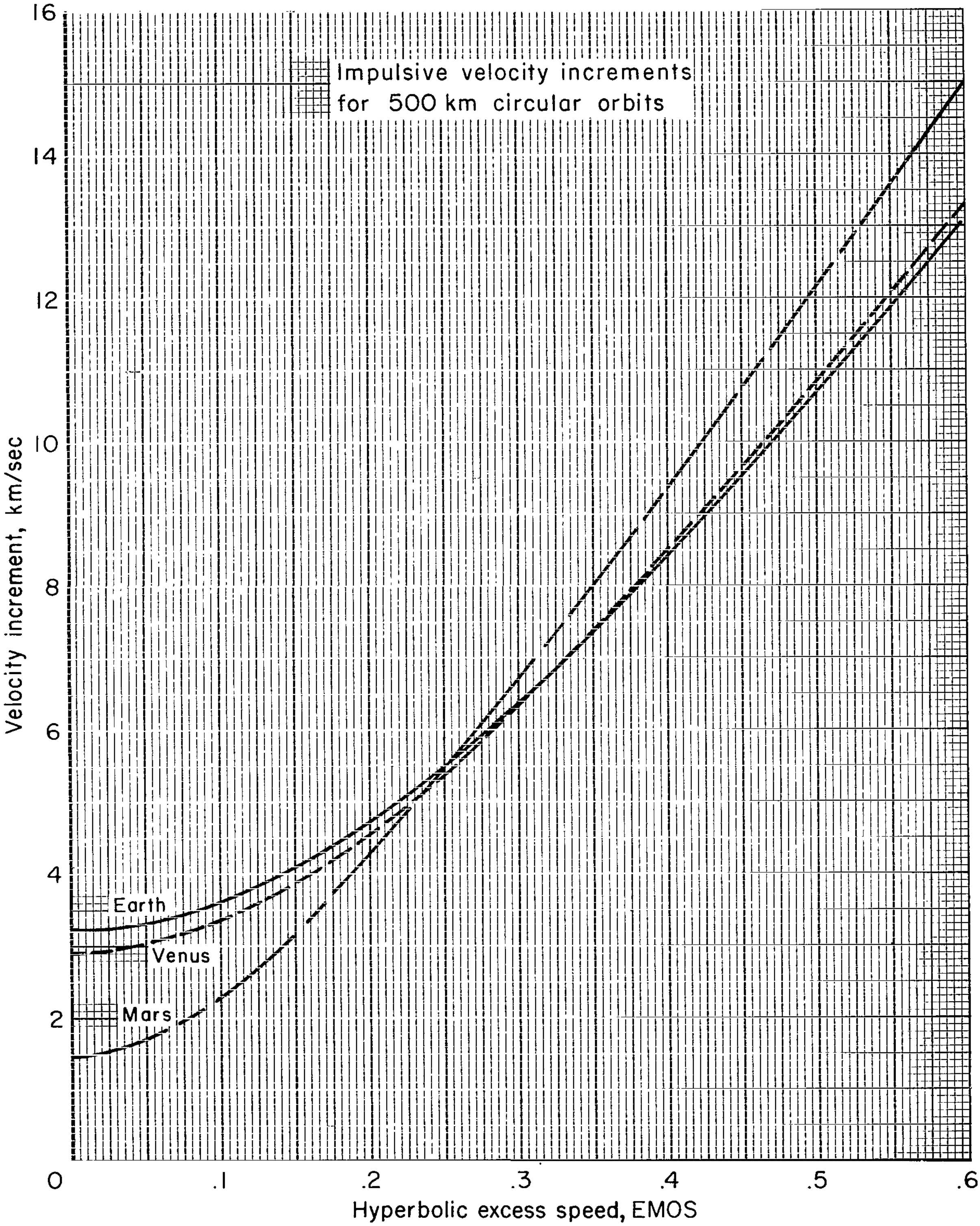


Figure 6. – Impulsive velocity increments (Earth mean orbital speed (EMOS) = 29.8 km/sec).

TABLE 1.– MISSION MODE COMPARISON; 30 DAY STOPOVERS

(a) Propulsive capture at the planet

Mission	Selection Criterion	Trip Time	$\Sigma\Delta V$ (a)	$\Delta V_{LV\oplus}$	$\Delta V_{AR♂}$	$\Delta V_{LV♂}$	$V_{AR\oplus}$	Lv⊕ (b)	Pass♀	Ar♂	Lv♂	Pass♀	Ar⊕	Peric (c)
1980 Outbound Swingby	Nominal[d]	500	11.81	4.30	4.19	3.33	14.13	3840	3998	4140	4170		4340	1790
	Minimum energy[e]	680	10.23	4.30	3.40	2.53	14.72	3840	4002	4170	4200		4520	2160
1982 Inbound Swingby	Nominal	520	11.70	4.13	2.68	4.89	13.97	4990		5210	5240	5383	5510	7620
	Minimum energy	620	10.49	3.65	2.33	4.52	12.17	4930		5220	5250	5391	5550	4860
1984 Inbound Swingby	Nominal	460	12.98	3.64	4.61	4.73	12.54	5730		5910	5940	6076	6190	4560
	Minimum energy	600	12.25	4.43	3.36	4.46	11.60	5650		5910	5940	6096	6250	2100
1984 Direct	Nominal	400	12.97	4.07	3.57	5.33	16.14	5770		5930	5960		6170	
	Minimum energy	460	11.73	3.65	3.09	4.99	16.25	5740		5940	5970		6200	
1986 Outbound Swingby	Nominal	500	11.10	4.12	4.25	2.73	12.95	6160	6317	6500	6530		6660	4700
	Minimum energy	600	10.09	3.97	3.71	2.41	12.06	6150	6317	6520	6550		6750	4930
1986 Direct	Nominal	400	12.28	3.80	3.28	5.20	15.23	6560		6700	6730		6960	
	Minimum energy	460	11.17	3.58	2.67	4.92	16.30	6540		6720	6750		7000	
1988 Inbound Swingby	Nominal	480	10.80	3.99	2.52	4.29	15.04	7370		7520	7550	7734	7850	8010
	Minimum energy	560	10.06	3.75	2.36	3.95	12.07	7340		7520	7550	7746	7900	3450
1990 Outbound Swingby	Nominal	460	13.89	4.05	5.61	4.23	11.58	7840	7984	8110	8140		8300	1600
	Minimum energy	640	11.08	3.90	4.01	3.17	13.98	7830	7980	8140	8170		8470	7780
1990 Direct	Nominal	400	14.62	4.75	4.45	5.42	19.80	8160		8290	8320		8560	
	Minimum energy	460	13.49	4.61	3.88	5.00	19.74	8100		8280	8310		8560	
1993 Outbound Swingby	Nominal	520	11.72	4.36	4.47	2.89	13.16	8520	8670	8820	8850		9040	6660
	Minimum energy	600	10.98	4.18	4.49	2.31	11.47	8510	8668	8810	8840		9110	6520
1995 Inbound Swingby	Nominal	500	11.62	3.91	3.38	4.33	14.71	9670		9880	9910	10051	10170	2650
	Minimum energy	560	11.18	3.84	3.35	3.99	12.23	9660		9880	9910	10059	10220	1300
1997 Direct	Nominal	420	13.57	4.52	3.60	5.45	19.80	10460		10640	10670		10880	
	Minimum energy	480	12.43	3.62	3.10	5.71	19.36	10420		10650	10680		10900	
1999 Outbound Swingby	Nominal	480	12.93	4.36	4.51	4.06	15.01	10850	11004	11170	11200		11330	1420
	Minimum energy	680	10.01	4.19	3.28	2.53	14.76	10840	11006	11220	11250		11520	1910
1999 Direct	Nominal	400	13.35	4.22	3.99	5.14	16.79	11230		11390	11420		11630	
	Minimum energy	460	12.09	3.67	3.22	5.20	16.92	11200		11410	11440		11660	
1983 Venus	Nominal	400	12.75	3.85	4.15	4.75	13.87	5510		5610	5640		5910	
	Minimum energy	480	11.68	3.61	3.74	4.33	14.35	5505		5635	5665		5985	

[a]All speeds in km/sec.
[b]All dates JD-2440000.
[c]Pericenter altitude at Venus (km).
[d]Mission selected on basis of (Trip time x $\Sigma\Delta V)_{min}$.
[e]Mission selected on basis of $(\Sigma\Delta V)_{min}$

TABLE 1.– MISSION MODE COMPARISON; 30 DAY STOPOVERS – Concluded

(b) Atmospheric capture at the planet

Mission	Selection Criterion	Trip Time	$\Sigma\Delta V$ (a)	$\Delta V_{LV\oplus}$	$V_{AR♂}$	$\Delta V_{LV♂}$	$V_{AR\oplus}$	Lv⊕ (b)	Pass♀	Ar♂	Lv♂	Pass♀	Ar⊕	Peric (c)
1980 Outbound Swingby	Nominal[d]	500	7.10	4.31	8.87	2.79	13.25	3840	3996	4120	4150		4340	800
	Minimum energy[e]	640	6.48	4.26	8.88	2.23	12.61	3830	3997	4120	4150		4470	679
1982 Inbound Swingby	Nominal	540	8.64	3.88	6.25	4.76	13.20	4980		5200	5230	5384	5520	5780
	Minimum energy	620	8.16	3.65	5.77	4.52	12.17	4930		5220	5250	5390	5550	4870
1984 Inbound Swingby	Nominal	460	7.69	3.81	9.57	3.88	12.49	5730		5890	5920	6078	6190	6960
	Minimum energy	480	7.66	3.85	9.88	3.81	11.56	5750		5890	5920	6089	6230	2770
1984 Direct	Nominal	380	8.34	4.01	9.97	4.33	15.68	5760		5890	5920		6140	
	Minimum energy	420	8.02	3.79	9.67	4.23	15.81	5740		5890	5920		6160	
1986 Outbound Swingby	Nominal	520	6.08	3.97	9.60	2.11	12.24	6150	6315	6470	6500		6670	2590
	Minimum energy	580	5.94	3.95	9.61	1.99	11.64	6140	6315	6470	6500		6720	2400
1986 Direct	Nominal	380	8.20	4.02	9.68	4.18	14.39	6550		6660	6690		6930	
	Minimum energy	420	7.99	3.93	9.02	4.06	14.52	6520		6660	6690		6940	
1988 Inbound Swingby	Nominal	480	8.24	4.03	6.24	4.21	14.96	7370		7510	7540	7733	7850	7920
	Minimum energy	560	7.61	3.78	6.01	3.83	12.06	7340		7510	7540	7745	7900	3080
1990 Outbound Swingby	Nominal	460	7.69	4.00	9.95	3.69	11.75	7830	7979	8100	8130		8290	2150
	Minimum energy	600	6.78	3.98	9.88	2.81	12.44	7820	7975	8100	8130		8420	3510
1990 Direct	Nominal	400	9.75	4.78	9.90	4.97	18.38	8140		8260	8290		8540	
	Minimum energy	440	9.28	4.53	8.38	4.75	19.68	8120		8270	8300		8560	
1993 Outbound Swingby	Nominal	540	6.87	4.18	7.93	2.69	12.53	8510	8668	8810	8840		9050	6530
	Minimum energy	620	6.39	4.15	8.14	2.25	11.45	8500	8667	8800	8830		9120	6130
1995 Inbound Swingby	Nominal	500	7.86	4.03	8.33	3.83	14.47	9670		9850	9880	10049	10170	2710
	Minimum energy	560	7.46	4.00	8.22	3.46	12.26	9660		9850	9880	10056	10220	1410
1997 Direct	Nominal	400	8.97	4.25	9.65	4.72	18.60	10450		10590	10620		10850	
	Minimum energy	440	8.57	3.93	9.25	4.64	18.75	10420		10600	10630		10860	
1999 Outbound Swingby	Nominal	480	7.92	4.36	8.54	3.56	14.89	10850	11003	11160	11190		11330	980
	Minimum energy	640	6.29	4.19	8.54	2.10	13.55	10840	11003	11160	11190		11480	840
1999 Direct	Nominal	400	8.28	3.85	9.56	4.43	16.49	11210		11360	11390		11610	
	Minimum energy	420	8.10	3.78	9.44	4.32	16.53	11200		11360	11390		11620	
1983 Venus	Nominal	380	8.35	4.15	13.86	4.20	13.54	5505		5585	5615		5885	
	Minimum energy	460	7.98	3.66	11.50	4.32	14.00	5490		5610	5640		5950	

[a]All speeds in km/sec.
[b]All dates JD-2440000.
[c]Pericenter altitude at Venus (km).
[d]Mission selected on basis of (Trip time x $\Sigma\Delta V)_{min}$.
[e]Mission selected on basis of $(\Sigma\Delta V)_{min}$.

such as planetary encounter dates, velocity requirements for circular capture orbit, and entry speeds are shown. Both the minimum energy and the nominal missions are listed for each opportunity.

Elliptical Capture Orbits

The velocity increments required at Venus and Mars for arrival or departure are subject to considerable variation depending on the actual maneuvers performed. For example, maneuvers performed at pericenter of an eccentric orbit would reduce both the arrival and departure velocity increments since the orbital pericenter velocity increases noticeably with eccentricity. Figure 7 shows the velocity reductions as a function of orbit eccentricity. The MEO's shown assume that arrival and departure maneuvers are performed at periapsis.

The velocity reductions are approximately a linear function of eccentricity. Thus, the variation in MEO with eccentricity is also nearly linear; requirements for any eccentricity can easily be approximated by interpolating between the requirements for a circular orbit and eccentric orbit. In this document, an eccentricity of 0.7 is chosen for both Mars and Venus. A substantial portion of the maximum velocity reductions are thus realized and result in savings of 1.0 and 2.2 km/sec at arrival or departure for Mars and Venus, respectively. At the same time, the orbital periods for elliptic orbits remain relatively short compared to the total stopover times – 11 hr at Mars and 10 hr at Venus.

It should be pointed out that for any specific mission neither arrival nor departure will occur at pericenter of the orbit. Recent studies indicate, however, that maneuvers carried out as much as 30° in true anomaly from pericenter result in increases in incremental velocity of about 10 percent for an eccentricity of 0.7. Maneuvers conducted within this range of true anomalies are generally adequate to satisfy the geometrical constraints at both arrival and departure. Thus, the results contained herein, while somewhat optimistic from this standpoint, are nevertheless representative.

Launch Delay Effects

The velocity requirements discussed previously do not include any launch window penalties from Earth orbit or planet orbit. Such penalties must be considered in the context of the orbit operations since they depend highly on the particular parking orbit characteristics. Such factors as orbit inclination, pericenter altitude, and eccentricity have a significant effect on the additional velocity increments needed to satisfy launch delay requirements. These velocity penalties could have been determined, of course, had particular orbital characteristics been assumed. In this document, however, the details of the planet operations are separated from the analysis of the total mission, and therefore no launch window penalties are included.

The rather complete analysis of launch window penalties for circular orbits contained in reference 5 indicates that a velocity penalty of only about 10 percent of the coplanar departure ΔV is sufficient to obtain as long as a 60-day launch window at Mars. At Earth and Venus the same penalty is required to obtain a 20-day launch window. These penalties are predicated on the use of a three-burn maneuver at departure from a circular orbit. The first burn occurs at pericenter and increases the orbit eccentricity to about 0.9; the second burn occurs near

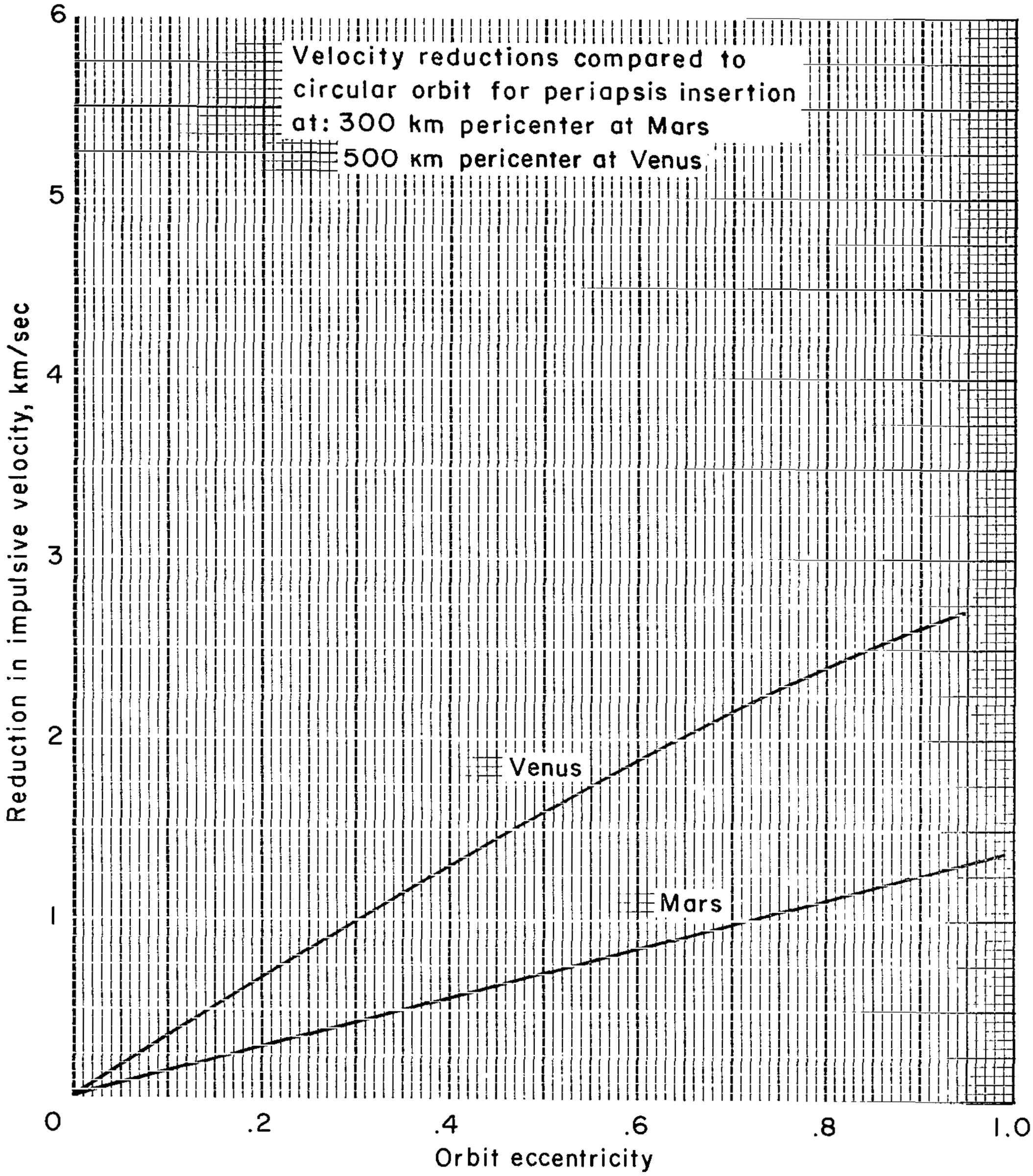


Figure 7.– Velocity reduction for entering or leaving elliptical orbits.

apocenter of the intermediate ellipse to rotate the parking orbit plane so that it contains the departure asymptote; the third occurs near pericenter and provides the required departure velocity.

Although elliptic capture orbits do not lend themselves to generalized conclusions due to the interdependence of the arrival and departure maneuvers, reference 6 indicates that 5 percent of the total ΔV (arrival and departure) generally will yield a launch window of up to 50 days at Mars for most stay times. At Venus it is not unreasonable to expect velocity penalties for elliptic orbits of about half the circular orbit requirements.

The inclusion of the planet departure velocity penalties mentioned above would increase the MEO values shown in figures 3 and 4 by 5 to 10 percent. The inclusion of the quoted Earth departure penalty would result in another penalty of about 10 to 15 percent.

A single propulsion stage that performs both the arrival and departure maneuvers at the destination planet merits consideration as one means of providing for the effects of launch delays. The operational advantage of employing a single stage is that it increases the performance margins at planetary departure. Margins must be available for both arrival and departure, but if separate stages were employed and a nominal arrival took place, the unused propellant would be wasted. By using a single stage, all unused arrival propellant would be available for the departure maneuver. Such a stage would be somewhat larger than necessary, compared to two optimized stages, if arrival and departure took place on time. If, for example, a single stage space-storable propulsion system were employed, the MEO values shown in figures 3 and 4 would increase by about 8 percent for a circular orbit and about 3 percent for an elliptic capture orbit. Although the propellant tank would also be larger than necessary if a nuclear system were employed the weight saved by the elimination of one of the engines actually renders the single-stage concept optimum from an MEO standpoint. The MEO would be reduced by 2 and 5 percent for circular and elliptic capture orbits, respectively.

SCALING LAWS

Recommended weight scaling laws, other system characteristics, and environmental considerations are presented in this section. These data, together with various analytical simplifications made to expedite the computational procedures, were used in developing the MEO requirements given in figures 3 and 4.

The data reflect technology and knowledge of the environment circa 1975, which in turn reflects the projected operational capability of the early 1980's. Because of this projection, it is impossible to justify all results and the attendant assumptions. About half the results are based on analyses conducted within the Mission Analysis Division; the remainder were obtained from other NASA and industry sources. The background data and system descriptions are given in sufficient detail to permit interpretation of the results with a minimum of effort.

Environmental Factors Affecting System Requirements

Thermal protection— The spacecraft attitude can be controlled fairly precisely with only a modest expenditure of propellant. Therefore, it is assumed that during transit, the longitudinal axis of the spacecraft is oriented toward the Sun with the mission module on the sunward end of the spacecraft. The average solar incidence angle on the side walls of the propellant tanks is taken to be 5°. A uniform steady-state temperature on the tank surface is assumed. Last, it is assumed that the absorptivity to emissivity ratio (α/ϵ) of the tank surface is 0.20, a value lower than currently available but higher than optimistic projections for future surface coatings.

The received solar thermal energy was computed on the basis of the time-average solar distance for each leg of the distinct mission profiles. Representative values were computed from the nominal 1980 outbound swingby, 1982 inbound swingby, and 1984 direct Mars missions, and the Venus stopover mission contained in table 1. Surface temperatures in planetary orbit, which include reradiation from the planet, were obtained from various sources and are predicated on a vehicle attitude that is horizontal relative to the planet surface.

On the basis of steady-state temperatures, and under the above assumptions, the uniform temperatures over the entire tank surface are as shown in table 2.

TABLE 2.– SURFACE TEMPERATURES, °K

Mission phase \ Mission	Venus stopover	Mars direct	Mars outbound swingby	Mars inbound swingby
Inbound leg	106	112	94	116
Planetary orbit	250	195	195	195
Outbound leg	116	101	117	96
Earth orbit	220	220	220	220

An optimum relationship can be established between insulation weight, propellant allowed to vaporize, and internal tank pressure when a particular mission and design is being analyzed. In lieu of this detailed optimization, it is assumed here that all heat to the propellant results in boiloff. Under these assumptions, the insulation weight for jth propulsion stage, which minimizes initial mass in Earth orbit (e.g., ref. 7), is:

$$W_{ins_j} = A_j \sqrt{\frac{\rho k(1 + c) \sum_{i=1}^{j} (\theta \Delta T)_i}{h\mu_j\mu_{j-1} \cdot \cdot \cdot \mu_1}} \tag{1}$$

The corresponding propellant boiloff for the jth stage is

$$W_{bo_j} = W_{ins_j}(\mu_j\mu_{j-1} \cdot \cdot \cdot \mu_1) \tag{2}$$

The parameters are defined as follows:

A_j total tank surface area, including forward and aft bulkheads (assumed hemispherical)

ρ insulation density

k thermal conductivity of insulation

ΔT temperature difference between tank and propellant

θ exposure time

h propellant heat of vaporization

c heat leak parameter (see below)

i index indicating each previous phase of the mission where i = 1 (Earth orbit); i = 2 (outbound leg); i = j (mission phase immediately preceding ignition of the jth propulsion stage)

μ_j mass ratio (initial mass/burnout mass) of the jth stage[4]

The heat input to the propellant is based on the use of NRC-2 insulation (ref. 8) with a density of 0.048 gm/cc (3.0 lb/ft^3) and a conductivity of 1.072×10^{-8} kcal/sec-m-°K (2.6×10^{-5} Btu/hr-ft-°R). Temperature differences and heats of vaporization are based on propellant characteristics discussed later.

In addition to the heat input from the sidewall temperatures of table 2, a heat input through the forward end of the planetary departure stage is considered since it is adjacent to the mission module whose interior is at room temperature of 294° K. The insulation thickness of this forward bulkhead is determined in the above manner with A_j representing the bulkhead area and a ΔT corresponding to the 294° K temperature.

The effect of heat leaks into the tank can be estimated parametrically by inclusion of the constant c, where c = 0 implies no heat leaks. The MEO values in figures 3 and 4 are based on a value of c = 1 (heat leaks equivalent to the heat transfer through the insulation). This assumption should not be construed either as a recommendation or as a design criterion. Until sufficient data are available, any value of c between 0.1 and 10 could probably be justified.

Specific values of propellant boiloff and insulation weights clearly depend on the particular mission, propulsion stage, and propellant combination. However, the following general results will place the influence of the requirements for thermal protection in some perspective. The optimum insulation weight corresponds to approximately 5 percent of the weight of the tank plus engine. The corresponding propellant boiloff weight of each stage varies between 2,000 kg and 5,000 kg.

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[4] If the insulation of the jth stage is jettisoned prior to startup the optimum insulation weight is independent of μ_j. Thus in the above equations, set $\mu_j = 1$. Similarly, if the propellant tanks are topped in Earth orbit (or if the time spent in orbit is small) set $\mu_1 = 1$ for the Earth departure stage.

These general results are independent of mission or system characteristics. It should be pointed out that the MEO values in figures 3 and 4 include the propellant that would have boiled off during a 30 day period in Earth orbit.

Meteoroid protection— Many uncertainties surround the meteoroid protection requirements for space vehicles both because of a lack of knowledge concerning the environment and because of incomplete experimental data concerning impact phenomena. Nevertheless, it is appropriate to establish flux and penetration relations so that the meteoroid protection analysis can be treated in a consistent manner.

Flux models: The following flux models are suggested:

Cometary (refs. 9 and 10)

$$\log \Phi_c = -14.49 - 1.213 \log m - 2 \log r \tag{3}$$

Asteroidal (ref. 10)

$$\log \Phi_a = -31.01 + 23.74\, r - 8.635\, r^2 - 0.84 \log m \tag{4}$$

where

Φ	number of particles/m^2-sec
m	particle mass, gm
r	heliocentric radius, au

Additional environmental characteristics are:

$$\rho_p = \text{particle density; 0.5 gm/cc for cometary, 3.5 gm/cc for asteroidal} \tag{5}$$

$$V = \text{impact velocity; } 20\, r^{-1/2} \text{ km/sec for cometary (isotropic), } 10\, r^{-1/2} \text{ km/sec for asteroidal (unidirectional)} \tag{6}$$

The cometary flux model does not differ significantly from the Whipple (1963) model and is generally considered to be the best current estimate of the near-Earth environment.

The asteroidal flux model is based on data from reference 10. The simplified model presented here does not consider explicitly the effects of spacecraft heliocentric longitude and latitude on the flux nor does it consider the effect of the asteroid's orbit eccentricity. However, the maximum effect of these parameters was applied in the development of the simplified model so that the flux model used herein is somewhat conservative. The assumptions do not appear to critically affect the flux values obtained since the most important variation is a function of heliocentric distance. The flux model given is most accurate in the 1-au to 1.6-au region. A curve fit is employed in the computations and irregularities in the function near the Mars orbit do not allow a good fit over the entire asteroid belt region. For missions to Mars and Venus, of course, this is of no consequence.

Penetration models: Of the many penetration models which have been proposed by various investigators, the following model (ref. 11) is suggested as representative for single-sheet meteoroid bumpers:

$$\frac{t}{d} = 0.44 \left(\frac{1}{\varepsilon}\right)^{1/18} \left(\frac{\rho_p}{\rho_t}\right)^{1/2} V^{8/7} d^{1/18} \tag{7}$$

$\frac{t}{d}$	bumper thickness to particle diameter ratio
d	particle diameter, cm (particle assumed spherical)
ρ_p	particle density, gm/cc
ρ_t	bumper density, gm/cc
V	impact velocity, km/sec
ϵ	bumper ductility, elongation, cm/cm

Multiple-sheet bumper efficiency: Numerous investigations have shown that the use of multiple-sheet bumpers can improve the efficiency of meteoroid shield structures. Efficiency increases with increasing sheet spacing and it also appears that further improvement can be obtained by use of a low-density absorbing medium. Since experimental programs are incomplete, however, specific values of efficiency are open to conjecture. Until firm data are obtained, the somewhat conservative values of reference 12 are adopted:

$$\bar{t} = K_1 T \tag{8}$$

where

$\bar{t}$	total thickness of double sheet
T	equivalent single-sheet thickness

The efficiency factor K_1 varies between 0.20 and 0.50 depending on the spacing between the sheets and type of filler material.

Shielding requirements: Single-sheet thickness requirements for each of the four distinct mission profiles were computed by integrating the sum of the cometary and asteroidal fluxes along the trajectory. The probability of no penetrations was fixed at 0.99. Due to the isotropic nature of the cometary flux, the total area represents the potential target area for cometary particles. Since the asteroidal particles will approach the vehicle from a predictable direction, however, the projected area of the tank wall (*l*d) was used in the determination of the probability of impact. The required thickness was then assumed to be applied to the entire surface area of the vehicle to permit total flexibility in the vehicle attitude. Finally, an efficiency factor of 5 (K_1 = 0.2) was assumed.

In order to obviate the need for rigorously computing the shield thickness for every mission and surface area, the actual values used in the MEO calculation were estimated from the relationship:

$$\bar{t}_{act} = \bar{t}_{ref}\left[\frac{(A\Delta T)_{actual}}{(A\Delta T)_{ref}}\right]^{1/3} \tag{9}$$

where A is the surface area and ΔT is the exposure time. A reference area of 400 m^2 was used, and reference exposure times of 500 days (Earth arrival), 250 days (planet departure and arrival) and 30 days (Earth departure) were employed. Surface density requirements for the meteoroid shield based on values of $\bar{t}_{ref}$ are shown in table 3 for all spacecraft modules. Each value reflects the total requirement. For example, a value at Mars departure includes the shielding required for the exposure in Mars orbit, during the outbound leg, and in Earth orbit. The shielding is assumed added to the basic spacecraft structure. The relatively minor contribution of the basic structure to the meteoroid protection is neglected in this analysis. It should also be pointed out that the ascent load-bearing shell, discussed later, could provide the shielding requirement while in Earth orbit. This approach was not employed, however, in the development of the MEO requirements. In the analysis, it was assumed that the shield of each propellant tank was jettisoned just prior to startup of the engine.

TABLE 3.– TOTAL DOUBLE–SHEET SURFACE DENSITY (GM/CM²)

Location \ Mission	Venus stopover	Mars direct	Mars outbound swingby	Mars inbound swingby
Earth return	0.99	1.24	1.71	1.41
Planet departure	.87	1.10	1.00	1.27
Planet arrival	.87	1.03	.72	1.23
Earth departure	.60	.60	.60	.60

Radiation protection– The shield weight required to protect the crew from solar proton radiation can represent a sizable fraction of the mission module weight and is therefore a subject of considerable study. The results presented here are based on a recent study (ref. 13) of this factor.

Solar events occur sporadically and vary greatly in proton fluxes and energy spectra. The dose received during the mission depends on the number of events encountered, their fluxes and spectra, and the shielding protection available. The dosage also depends on distance from the Sun. Since this relationship is not adequately understood, for convenience, the not unreasonable assumption is made that all dosages at any distance are equivalent to those at 1 au. Because of the irregular variations in flux, spectrum, and frequency, the only reliable method of predicting the radiation environment for long duration missions is by statistical evaluation of past data on solar proton events to produce distribution functions for the critical quantities. From such distributions, model environments may be generated and assigned appropriate probabilities of occurrence.

Shield requirements: With the dose per event distribution and the frequency distribution available, the mission-dose distribution was derived by taking a large number of samples. The number of events per mission was calculated, the dose per event distribution was randomly sampled, and the event doses summed to obtain the dose per mission. Figure 8 shows the dose as a function of thickness for an aluminum shield, based on a probability of 0.99 of not exceeding a given dose. Figure 9 shows similar data for a polyethylene shield. Since the requirements depend on the fluctuations in radiation that occur during the 11-yr solar cycle, the requirements for both solar minimum and an average solar maximum are shown.

From these data, it is possible to compute the shield weight requirements once other specifications are made. First, it is assumed that the total allowable skin dose per mission is 150 rem. Of this total, it is assumed that 20 rem per year accrue, independent of solar activity, from galactic background radiation. It is also assumed that 10 rem per year may accrue from onboard radiation. If neither nuclear propulsion nor nuclear auxiliary power systems are employed, this 10 rem dosage is taken as a contingency in the weight analysis. Thus, the remaining radiation may be allowed to accumulate from solar events.

Second, it is assumed that the Earth reentry module (ERM) will be used as a radiation storm shelter. While provisions for a separate shelter within the mission module (MM) could be made, the storm shelter approach is taken since the ERM provides necessary life support functions. Primary spacecraft maneuvers can be carried out by brief trips to the MM command center.

Figures 8 and 9 indicate that polyethylene is a more efficient shield material than aluminum. Thus, it is assumed that the ERM housing within the MM will be covered with polyethylene. The basic aluminum structure of the MM and ERM also afford some radiation protection. Representative thicknesses are as follows:

MM structure	1.22 gm/cm^2
MM meteoroid shield	1.00
ERM structure	3.90
ERM heat shield[5]	1.72
TOTAL	7.84 gm/cm^2 (16.1 lb/ft^2)

In the case of the ERM heat shield the thickness given represents the average distributed over the entire ERM surface. The dosage not attenuated by this inherent shielding must be attenuated by polyethylene. The necessary polyethylene surface density is shown in figure 10. The shield is applied to a surface area of 46 m^2. This is the surface area of an eight-man-crew Apollo-type configuration ERM designed for an Earth entry speed of 13.5 km/sec.

Since the onset and decay of solar maximum occurs rather abruptly, it is assumed that solar maximum and solar minimum each occupy half of the 11-yr cycle. The onset of solar maximum in the time period of interest was taken as early 1977. For missions that take place during the transition, a mean shield thickness was employed.

[5] For radiation shield purposes the heat shield material is assumed to afford the same protection as aluminum.

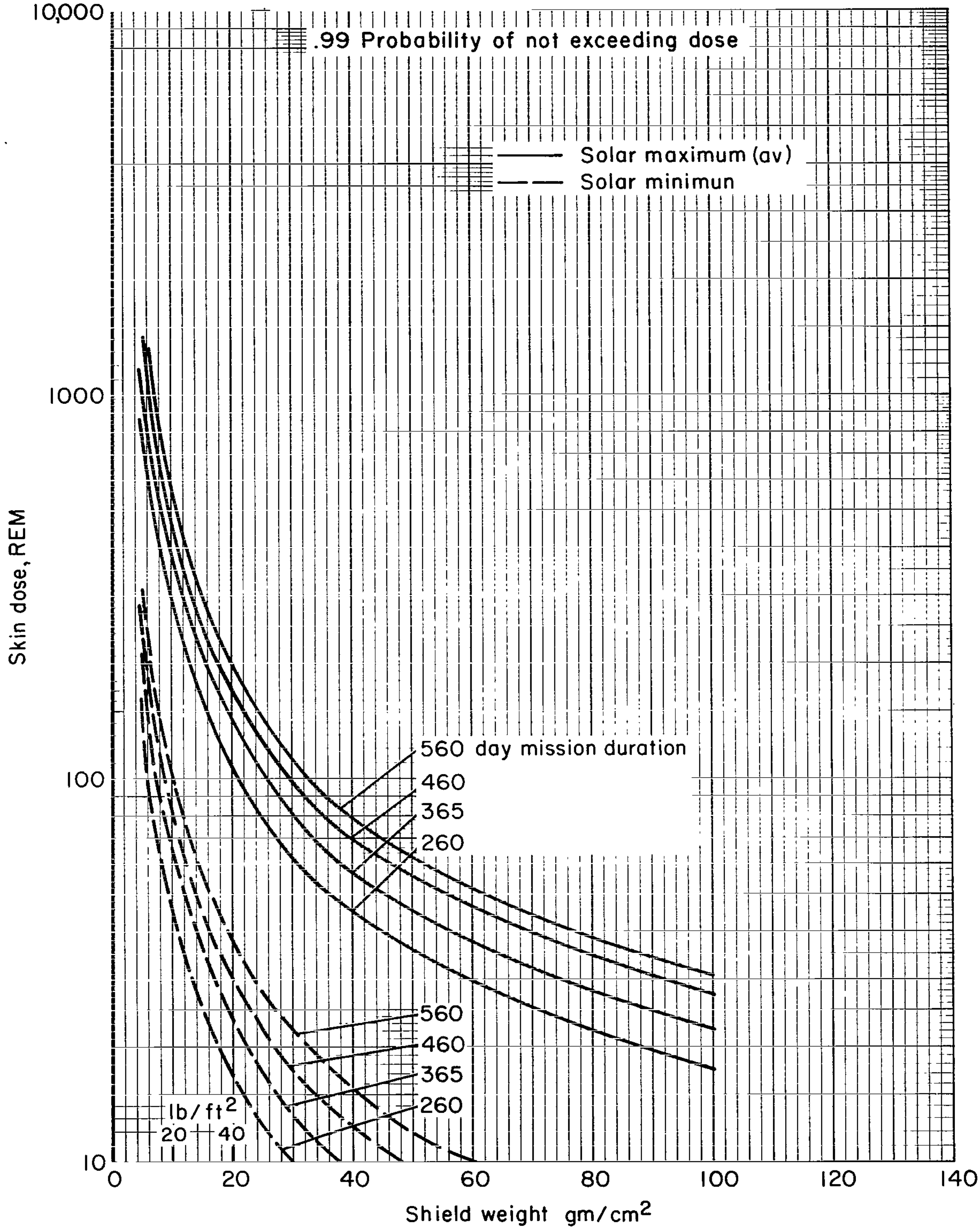

Figure 8.– Radiation shield weight – aluminum.

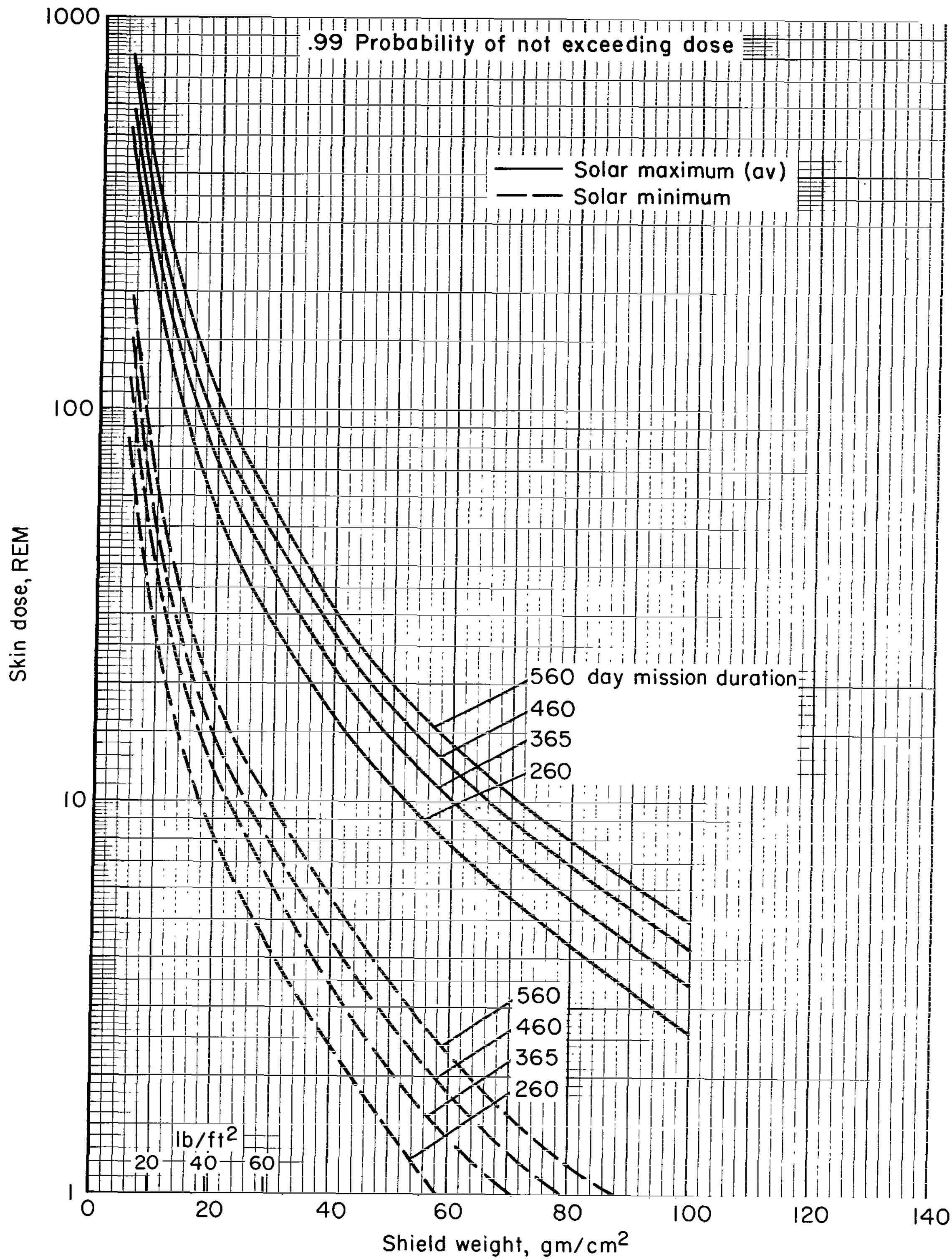

Figure 9.– Radiation shield weight – polyethylene.

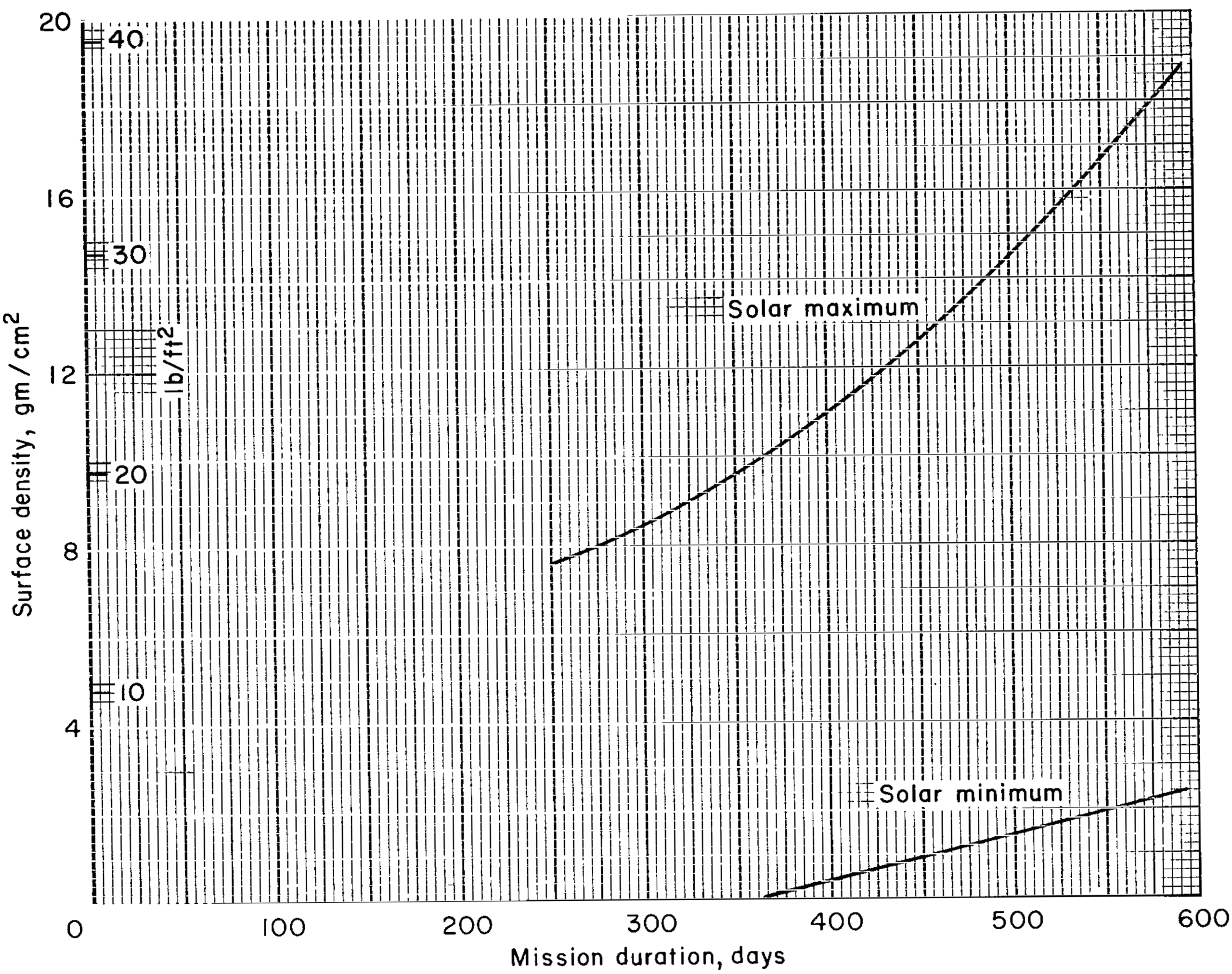

Figure 10.– Polyethylene shield requirements.

The solar maximum data of figure 10 were linearized to contain the points at mission durations of 250 days and 520 days (resulting in errors less than 10 percent) and the thickness was applied to the 46 m^2 area. Table 4 gives the radiation protection thus obtained in terms of weight of polyethylene as a function of mission duration T, in days.

TABLE 4.– RADIATION PROTECTION WEIGHT REQUIREMENTS (KG)

Solar maximum $W = -186 + 14.4T$	Solar minimum $W = -760 + 2.1T$	Solar mean $W = -472 + 8.2T$
	Mission	
1980 Outbound swingby 1988 Inbound swingby 1990 Outbound swingby 1990 Direct 1993 Outbound swingby 1999 Direct	1983 Venus conjunction[a] 1984 Inbound swingby 1984 Direct 1986 Outbound swingby 1986 Direct 1995 Inbound swingby 1997 Direct	1982 Inbound swingby 1999 Outbound swingby

[a]If a Venus mission were flown during solar maximum, the MEO requirements shown in figures 3 and 4 would increase by about 10 percent.

Manned Modules

The following paragraphs present parametric weight data for the mission module, Earth reentry module, and the Mars excursion module (including orbiter mission payloads). The particular details of the design philosophies and module weight breakdowns used to derive these scaling laws do not, of course, directly affect the MEO values. However, the details are given here to provide some rationale for the scaling laws as presented, and to make clear the "bookkeeping" system used in this work wherein certain subsystems such as attitude control or meteoroid protection are handled separately. Analysis of other designs, crew sizes, or weight breakdowns will lead to alternative module weights.

Mission module– The mission module is the command, control, experiment, and communications center for the spacecraft and serves as the living quarters for the crew for the entire mission. The spacecraft systems considered a part of the mission module are: crew support and environmental control, guidance and navigation, communications equipment and antennas, power supply, attitude control, and scientific and laboratory equipment. Of importance in the determination of MEO requirements is the gross weight of the mission module rather than the weight of each individual subsystem per se. As an aid to the interpretation of the overall weight scaling law for the module, however, a weight statement for a basepoint design is provided in table 5. This design is based on a partially closed ecology system (water and air recovery), a mission duration of 520 days and an eight-man crew; the sensitivity of MEO to crew size is about 4 percent per man. Each weight element indicated is believed to be representative of that subsystem, but the value shown is not based on a detailed analysis. Consequently, the weight breakdown must not be interpreted as a recommendation of subsystem characteristics.

TABLE 5.– BASEPOINT MISSION MODULE WEIGHT (EIGHT–MAN CREW)

Subsystem	Weight, kg
Crew and life support	7,250
Guidance and navigation	530
Communications	680
Scientific equipment	2,500
Airlock	230
Centrifuge (body conditioning)	230
Auxiliary power (10 kW at 500 kg/kW including conditioning and distribution)	5,000
Airlock usage and leakage at 2.3 kg/day	3,550
Repressurization every 90 days	2,370
Module structure	8,350
Repair and replacement (5 percent)	1,590
Contingency (5 percent)	1,590
Gross weight at Earth departure, not including meteoroid protection, radiation protection, attitude control system and ERM	33,870

The total weight required for supplies (e.g., food, repressurization, and personal equipment wearout) is 22 kg/day for an eight-man crew. The weight of the waste material expended from the mission module totals 13 kg/day, including airlock usage, leakage, and jettisoning of equipment at various points throughout the mission.

The most useful information in performance analyses is the mission module weight at Earth return at separation of the ERM. From the data in table 5 and assuming the supply rate is independent of mission duration, the mission module weight without supplies is 22430 kg (i.e., 33870 kg less the 11440 kg of supplies needed for a 520-day mission). The net change in module weight on a daily basis is an increase of 22 kg due to supplies, less the 13 kg of jettisonable items. The partial module weight at Earth return is thus

$$W = 22430 + 9T \text{ (kg)} \tag{10}$$

Based on a 10-m diameter and a free volume allowance of 21.2 m^3 per man, the MM cylindrical length excluding the hemispherical end bulkheads is 8 m. Thus the exposed cylindrical surface area is 250 m^2. Because of this small area (compared to a propellant tank) the required meteoroid shield weight is small and is, for practical purposes, independent of the mission profile or duration. Consequently, a representative shield surface density of 1.0 gm/cm^2 (see table 3) was applied to all MM resulting in a constant shield weight of 2500 kg.

As pointed out later, a typical attitude control gas requirement is 3.6 kg/day. Assuming a 30-percent inert weight fraction, the attitude control hardware weight can be approximated as:

$$W = 1.1T \text{ (kg)}$$

Adding the radiation protection (table 4), meteoroid shield and attitude control system hardware, the MM weight (excluding ERM) at Earth return is as follows:

$$\left.\begin{aligned} &\text{Solar maximum} && W_{MM} = 24740 + 24.5T \text{ kg} \\ &\text{Solar minimum} && = 24170 + 12.2T \text{ kg} \\ &\text{Solar mean} && = 24460 + 18.3T \text{ kg} \end{aligned}\right\} \tag{11}$$

Progressing backward through analysis of the mission to obtain the MM weight at Earth departure, the weight of the expendable supplies and attitude control gas totaling 16.6 kg/day must be added to the Earth return weight.

Among the various manned modules, MM weight is one of the most significant in its effect on MEO requirements. Typically, a 10-percent change in the module weight at Earth return results in a 4- to 6-percent change in MEO for the lander missions. For orbiter missions, the sensitivity is increased to 6 to 9 percent.

Earth reentry module– The ERM is a separate vehicle contained within the MM until separation prior to the entry maneuver. During other phases of the mission the ERM serves as the radiation storm shelter for the spacecraft. Since the vehicle must be easily accessible and also permit mobility within itself during the few days of occupancy necessary in the event of a solar flare, a free volume of 1.4 m^3 for each crew member is provided. The ERM does not depend on the mission module and has its own life support, guidance and navigation, auxiliary power, and communications systems as well as those systems such as the heat shield and recovery parachutes required of an entry vehicle. The life support (open ecology system) and auxiliary power supply (fuel cells) systems are adequate for 5 days of independent operation.

The weight requirements for three distinct vehicle types, each characterized by a unique aerodynamic configuration, have been determined – Apollo, biconic and cone segment. The vehicle gross weight as a function of Earth entry speed is shown in figure 11. Included in the weights is a 365-kg scientific payload. Point designs for these vehicle types are given in reference 14. Data from these designs were used to establish the scaling laws from which a parametric study of these vehicle characteristics (ref. 15) was conducted. Information from reference 15 was used to compute the weights shown.

As shown in the figure, the biconic configuration exhibits a lower weight sensitivity to entry speeds than does the Apollo shape. Having a sharper forebody cone angle, the radiation heating to the lower surface is reduced at the higher speeds. The cone segment vehicle is most attractive at the higher speeds. Since it has a high fineness ratio, the onset of high radiation heat transfer rates is delayed and much of the heating remains convective. At the lower speeds, however, this configuration is less attractive primarily because of its low volumetric efficiency.

For calculation of the MEO requirements in figures 3 and 4, only the Apollo and biconic vehicles were considered. There were two basic premises for this decision. First, the Venus swingby mission mode will most probably be employed whereas the utility of the direct mode is questionable. Second, only one ERM development program will be undertaken. As pointed out in the preceding section, with the exception of the 1997 opposition, entry speeds for the swingby

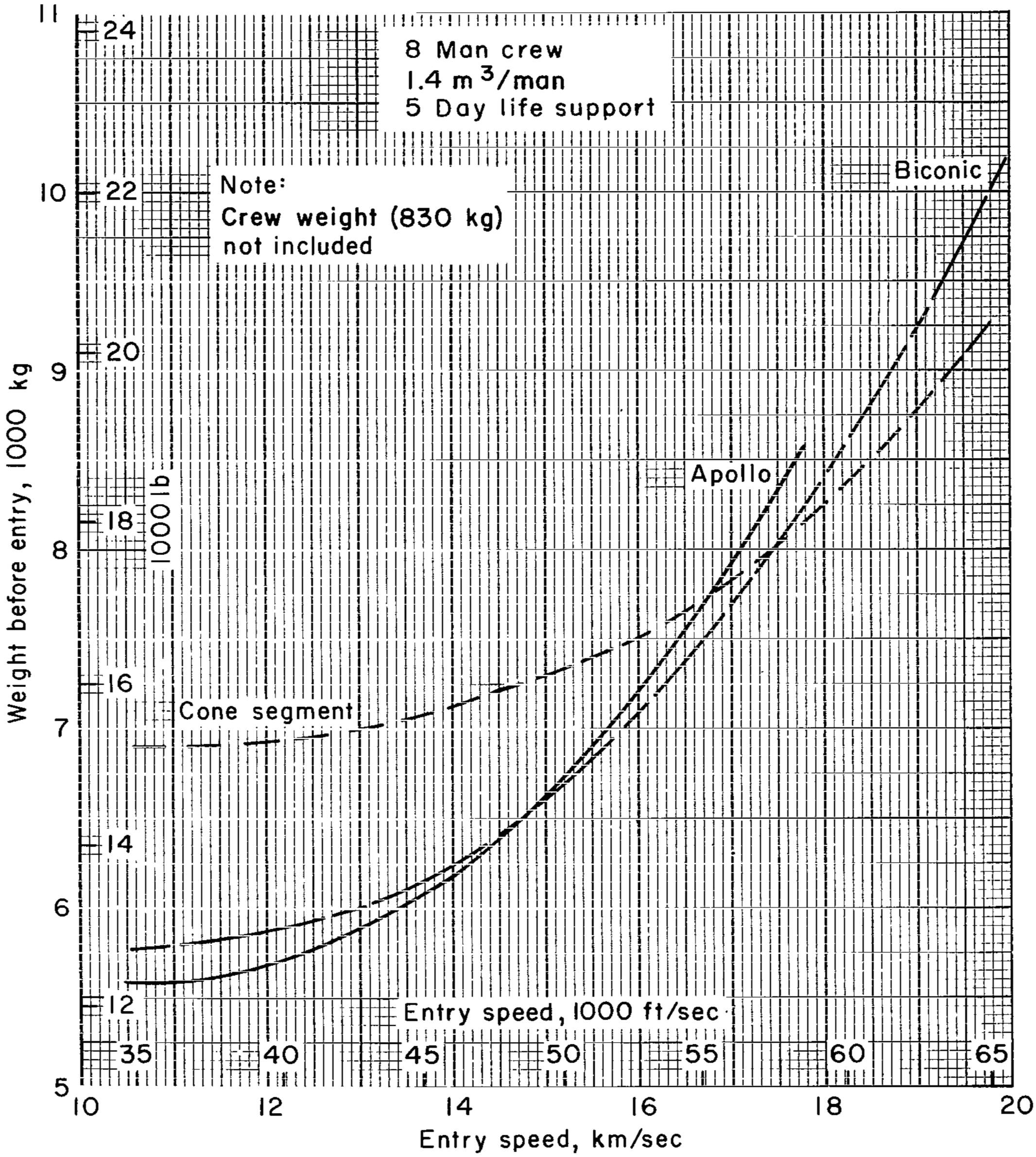

Figure 11.– Earth entry module weight.

mode and the Venus stopover can readily be constrained to lie under 14 km/sec. Thus, the Apollo configuration is desirable from an operational standpoint and will likely require the least development effort; for these reasons, this vehicle type is included in all swingby and Venus stopover MEO values. Conversely, no entry speed associated with the direct missions is less than about 15 km/sec; most are substantially higher, indicating another vehicle is appropriate. The cone segment is optimum at the highest speeds but is significantly heavier in the entry speed regime associated with the swingby mode. Therefore, the biconic vehicle was selected for the direct missions. Use of the biconic shape creates a modest weight penalty at high entry speeds, but the biconic vehicle is nearly optimum for a wide range of entry speeds. If the biconic vehicle were the single ERM to be developed, its influence on the remaining system characteristics at the low speeds would approximate those of the Apollo.

A unit weight increase of the ERM results in the same unit increase in MEO as does a unit increase in mission module weight. Because of its lower absolute weight, however, the percentage variation is significantly lower. A 10 percent change in ERM weight results in an MEO change of about 1 percent.

In addition to the change in MEO caused by a change in ERM weight, the effects on MEO of the use of propulsive capture of the ERM at Earth return as an alternative to aerodynamic entry is also pertinent. When a space-storable propulsion system is used to insert a crew module the size of the ERM into an elliptical 24-hr capture orbit, the MEO increases by 10 to 30 percent. These results apply to an entry speed range of 13 to 15 km/sec.

Mars excursion module and probes— A parametric design synthesis of Mars excursion modules (MEM) is given in reference 16. The primary results of that study are summarized in this section.

The MEM operations begin at separation from the spacecraft in Mars orbit. A small deorbit velocity increment is applied (at apocenter if an elliptical parking orbit is used) and aerodynamic deceleration takes place until terminal conditions are reached. Typically, the MEM velocity at this point is slightly greater than 1 km/sec. The main descent engine is then ignited and burns continuously until touchdown. The performance requirements include a 2-min hover. The low surface density of the Martian atmosphere (JPL VM-7 model atmosphere assumed) precludes the use of parachutes for the final descent of vehicles of the size under consideration.

Following surface operations, the descent systems are staged and the ascent vehicle follows a gravity turn trajectory to a burnout altitude of about 115 km followed by a coast and circularization into a 185-km phasing orbit. Finally transfer is made to the pericenter altitude of the mission module orbit, followed by the subsequent rendezvous. A 10-percent contingency is added to the ascent velocity requirements to provide for launch windows and rendezvous.

The design synthesis is predicated on an Apollo-type configuration. A crew size of four men is assumed with a surface stay time of 30 days. After rendezvous with the mission module, the ascent vehicle weight is approximately 2400 kg and includes a payload of 140 kg. This vehicle weight does not include the ascent propulsion system inert weight, although this inert weight was included in the MEO.

Two types of propulsion systems are considered-space storable (assumed $FLOX/CH_4$) and cryogenic (assumed $0_2/H_2$). The detailed characteristics of these propellant combinations are defined later. The specific impulse values are reduced for MEM applications to 383 sec and 433 sec, respectively, as an estimate of performance degradation that may arise because of engine throttling, restricted area ratios due to space limitations, etc.

Figure 12, based on data from reference 16, shows the gross weight of the MEM at separation from the mission module as a function of parking orbit eccentricity. As expected, the weight increases with eccentricity, primarily because of increased ascent velocity requirements. On the other hand the use of cryogenic propellants does not result in a MEM weight appreciably lower than that associated with the use of space-storable propellants. This relative inefficiency of the cryogenics is due primarily to the large volume required by the cryogenic propellants and the attendant increase in weight of the descent shell that surrounds the vehicle during entry and descent. The heavier weight of this descent shell, together with heavier propellant tanks themselves, tends to offset the higher specific impulse of the cryogenics. It is assumed that during the outbound leg, the MEM will be housed in a structure 10 m in diameter extending forward of the mission module for 8.5 m. This housing is intended to serve primarily as meteoroid protection. Based on a unit structural weight of 0.73 gm/cm^2, the housing weighs 2550 kg and is assumed to be jettisoned in Mars orbit.

It is assumed that the Mars landing missions will employ a rover vehicle of the type discussed in reference 17. This vehicle will transport two men in 2- to 5-day sorties for a range of about 800 km. The total landed weight including structures, expendables, and scientific equipment is 6950 kg. The gross weight prior to ejection from Mars orbit is also shown in figure 12. This weight decreases slightly with eccentricity because the deorbit maneuver takes place at apocenter and the vehicle is not returned to orbit. The description given here is not intended to be a recommendation of the characteristics of (or, necessarily, the need for) a rover vehicle. It is merely a documentation of the assumptions relevant to the MEO computations.

It is further assumed that 10,000 kg of scientific probes are deployed near the planet. These probes and the rover can be housed in a structure 4 m long during the outbound leg, the weight of the structure being 1,000 kg. For the orbiter missions, this 10,000-kg probe weight plus the 1,000-kg housing represents the entire weight off-loaded from planet orbit. In the Venus swingby mode, an additional 2,300 kg of probes are deployed at Venus passage.

The data summarized in figure 12 were used for the MEO calculations. For those missions using either nuclear or cryogenic propulsion for the arrival and departure maneuvers at Mars, cryogenic propulsion was considered for the MEM and rover. Similarly, for missions using storable systems for the major maneuvers, storable propulsion was used in the MEM and rover. Although this factor does not materially affect the MEO, it is important conceptually in that it reduces the number of propulsion systems to be developed.

The sensitivity of MEO to the weight off-loaded at Mars can be estimated by interpolation between the MEO values for lander and orbiter missions shown in figures 3 and 4. For the Venus orbiter mission, the sensitivity is indicated by presenting two different probe module weights, as discussed earlier.

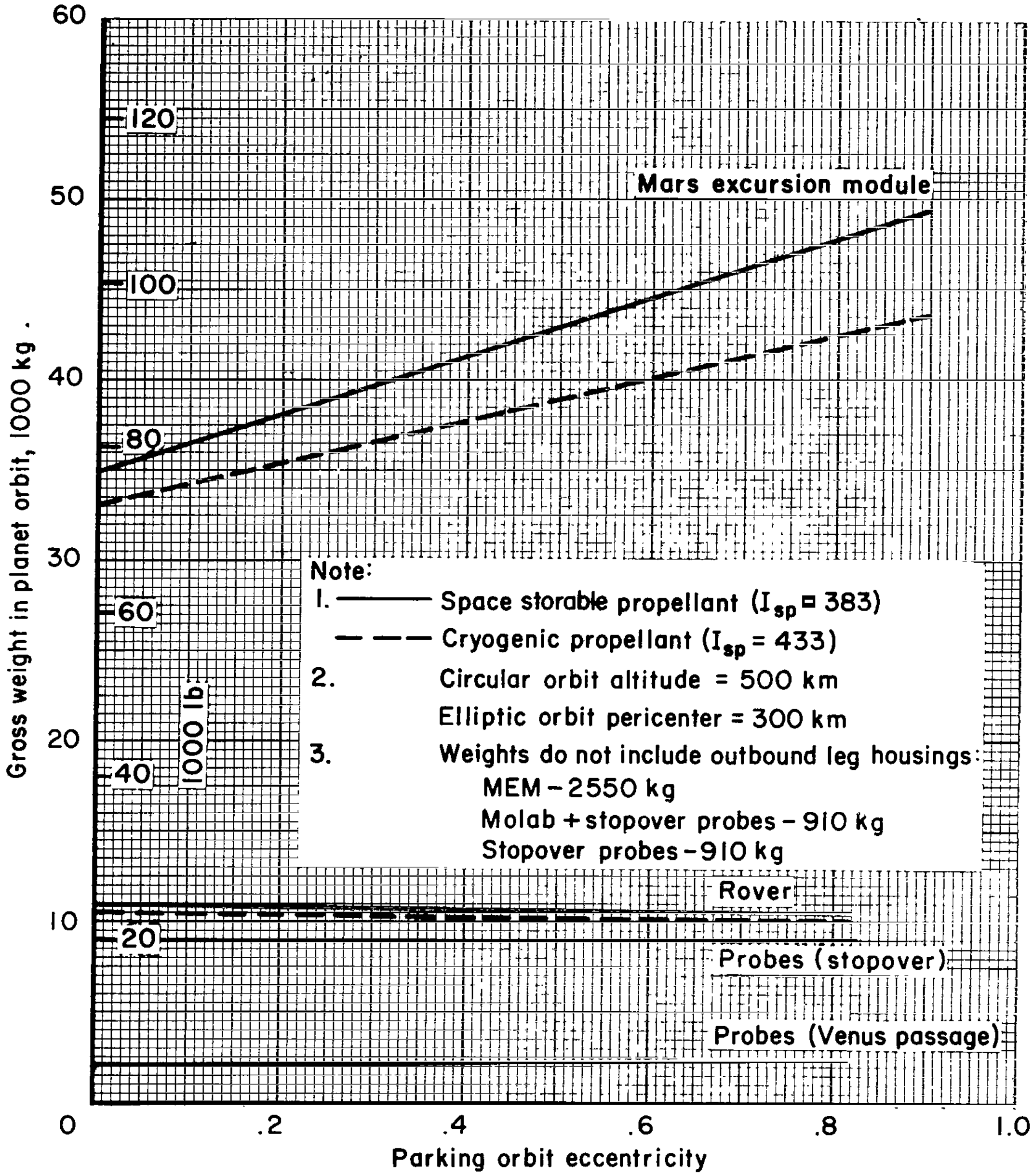

Figure 12.– Weight of systems off-loaded at planet.

Propulsion System Characteristics

The propellant and engine characteristics recommended for use in performance analyses are defined in table 6. The data were obtained from references 18 and 19 and from discussions with NASA personnel concerned with chemical and nuclear propulsion system technology development.

TABLE 6.– PROPULSION SYSTEM CHARACTERISTICS

Propellant	Delivered I_{sp}, vac.	Specific gravity	Heat of vaporization, kcal/kg	Boiling point °K at 1 atm	Engine		
					Thrust to weight	Thrust, kg	Weight, kg
N_2O_4/A50	330	1.21	99	292	50	–	–
FLOX/CH_4	405	1.06	43	86	100	–	–
O_2/H_2	450	0.318	108	20	100	–	–
H_2	850	0.0677	108	22[a]	–	34,000	10,500

[a]Pressure of 2 atm.

The last three systems in the table are representative of space-storable (S), cryogenic (C), and nuclear (N) propulsion systems, respectively. FLOX/CH_4 was chosen as the space-storable propulsion system for use in the MEO calculations rather than, for instance, FLOX/MMH or OF_2/MMH, because it will probably cost less to produce, the fuel and oxidizer are more nearly temperature compatible, and the specific impulse is slightly higher. The influence on MEO of using one of the other space-storable propellants, however, would be minor.

For the cryogenic system O_2/H_2 is recommended because it is inexpensive, it may be employed in some manner by the spacecraft life support system and it delivers a high specific impulse. Its only disadvantage relative to F_2/H_2, for example, is its low density.

N_2O_4/A50 is representative of Earth-storable propellants. This system was not used for any major maneuvers in calculations of the MEO requirements shown in figures 3 and 4. In all cases, however, it was used as the midcourse correction system.

The specific impulse values are based on the use of high chamber pressures and correspondingly large area ratios. The density values represent the weighted average for the propellant combination based on typical mixture ratios. The heat of vaporization and boiling point are shown for either the fuel or oxidizer, whichever has the lower boiling point (i.e., FLOX and H_2). The thermal analysis was carried out using these data for the propellant as a whole, rather than considering the fuel and oxidizer separately. The use of the lower boiling point results in modest conservatism.

The MEO requirements include the effects of optimum stage thrust-to-weight ratios. For the chemical systems, an optimized engine size and a 100:1 thrust-to-weight ratio were considered for each major propulsive maneuver. For the nuclear systems, the optimum number of engines of the given weight and thrust level was determined for each maneuver.

The most important performance characteristic of a propulsion system for manned planetary missions is, of course, the specific impulse. So long as the values used here are reasonably consistent with actual values of future systems, however, the actual MEO values will change only slightly from those quoted in this document. For instance, a change in specific impulse of 20 sec results typically in a 12-percent change in MEO for both chemical systems and a 5-percent change in MEO for the nuclear system.

The effects of the engine characteristics on MEO are less. For both the chemical and nuclear systems, a 20-percent change in engine thrust-to-weight ratio changes MEO by about 5 percent.

Structural Scaling Laws

Primary structure weights– Ideally, any weight scaling laws should be based on prior preliminary design studies. Moreover, these studies should consider various advanced materials and design concepts and should include a wide range of propellant loadings and propellant types. Whereas one recent study (ref. 20) fulfills some of these requirements for nuclear stages, no similar activity has been undertaken for chemical stages. Consequently, the approach taken here is first to establish the functional relationships among inert weight, propellant loading, and propellant type from the many chemical vehicles that have been produced or carried through a substantial development program. Finally, some effects of advances in technology and of designing vehicles expressly for interplanetary missions are estimated by including the nuclear stage point design with the current chemical vehicles to establish the recommended weight scaling law.

A compilation of chemical stage weight data, and a weight scaling law that agrees very well with these data, are given in reference 21. Figure 13 is a plot of $W_s\sigma^{0.533}$ as a function of $W_p{}^{0.9}$, where W_s is the weight of primary structure, σ is the propellant specific gravity and W_p is the total propellant weight. Note that the data points can be approximated quite well by a straight line, leading to the functional relationship

$$W_s = \frac{AW_p{}^{0.9}}{\sigma^{0.533}} + K \qquad (12)$$

where K reflects the minimum weight of the subsystems regardless of propellant loading. Note also that if the total propellant weight W_p is distributed evenly among n separate tanks, the total weight of all tanks becomes

$$W_{s_n} = \frac{An^{0.1}W_p{}^{0.9}}{\sigma^{0.533}} + nK \qquad (13)$$

where for current systems A = 0.194 and K = 500 if weights are in kilograms, and A = 0.21 and K = 1100 if weights are in pounds.

Since the data in figure 13 reflect a variation in σ of greater than 4 to 1, it is reasonable to expect that the above relationship will remain valid for another 4 to 1 variation (from O_2/H_2 to H_2). Superimposed on the figure (point 1) is the weight data for the nuclear module (with the same inclusions and exclusions that pertain to the chemical stages) for ascent to Earth orbit. The agreement with the average chemical data is remarkable – the computed value of A is 0.204.

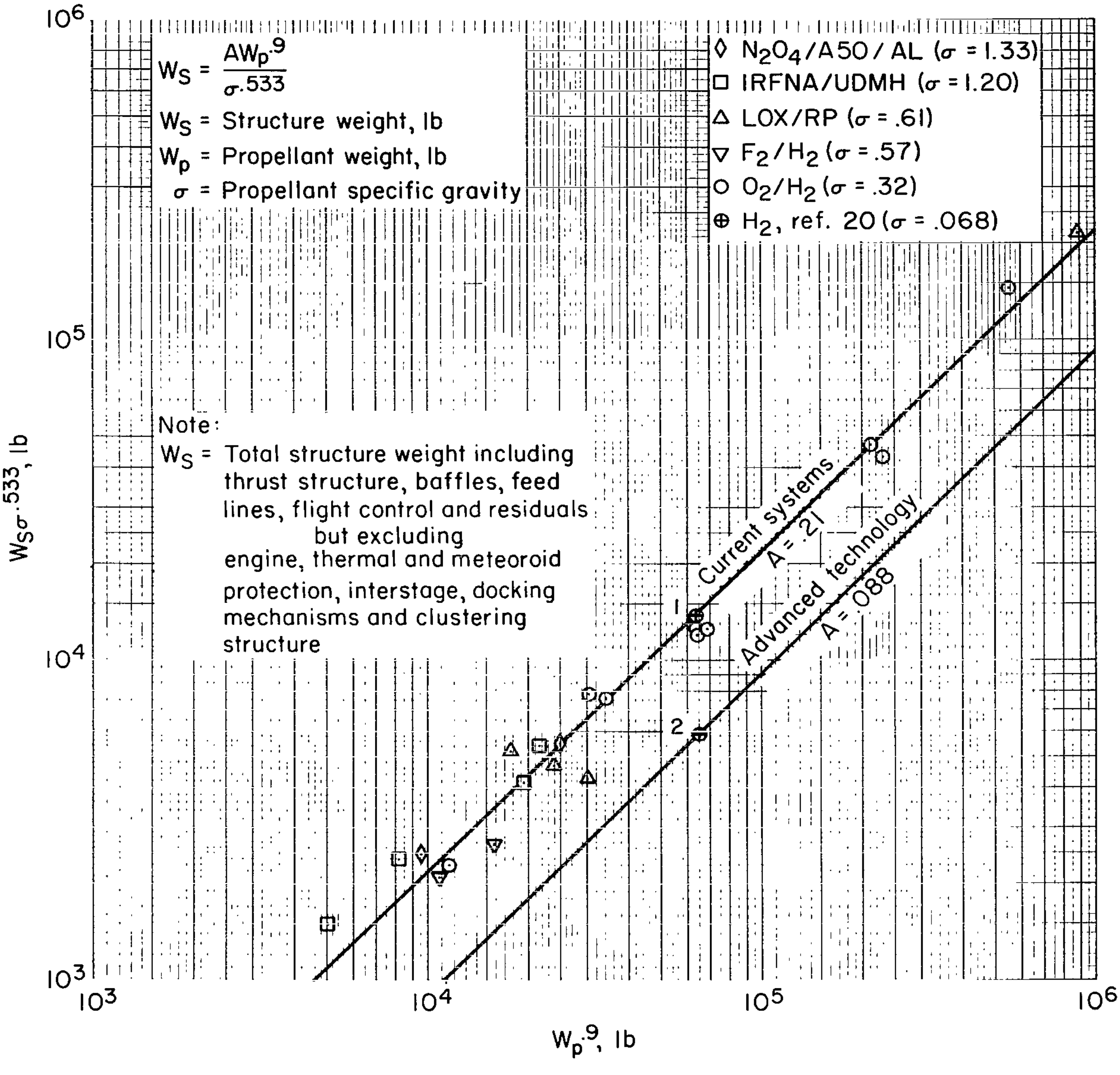

Figure 13.– Structural weight correlation for current systems.

Subsequent accelerations on the vehicle will, in general, be much less than 1 g in contrast to the ascent accelerations of several g's and the large bending moments at $q\alpha_{max}$. To take advantage of this situation, the ascent loads in the nuclear vehicle are borne by an external shell, which is jettisoned prior to departure from Earth orbit. This ascent load-bearing shell represents slightly more than half of the total stage structure. Figure 13 (point 2) shows the stage weight after the shell is jettisoned. The corresponding value of A is 0.088. This design concept can obviously be applied to chemical stages as well. While the savings in weight may not be as pronounced (for instance, heavier gauge materials may be needed to withstand higher internal pressures), they are assumed here to be of the same order.

More definitive studies are required to determine the utility of this concept. Meanwhile, the following weight scaling law for n tanks is recommended:

$$W_s = \frac{0.08n^{0.1}W_p^{0.9}}{\sigma^{0.533}} + 500n \text{ kg} \tag{14a}$$

$$W_s = \frac{0.09n^{0.1}W_p^{0.9}}{\sigma^{0.533}} + 1100n \text{ lb} \tag{14b}$$

The combined weight of the ascent load-bearing shells for the Earth departure, planet arrival, and planet departure stages is on the order of 50,000 kg regardless of spacecraft propulsion system. To determine the precise on-orbit payload requirements of the Earth ascent boosters, this weight should be added to the MEO requirements.

This design concept may be unattractive from the standpoint that the jettisoned material is large and heavy and could pose a serious threat to other Earth orbiting vehicles or perhaps to land masses. This is particularly true if multiple Earth orbit rendezvous of such vehicles is required. Other studies (ref. 22), however, have indicated that similar reductions in structural weight may also be achieved in the future by use of advanced materials (e.g., beryllium or titanium), better fabrication techniques (e.g., honeycomb or monocoque), and more rigorous analytical techniques to more accurately predict the structural loads. While these conclusions are based only on a study of large launch vehicles, it is reasonable to assume that the design of spacecraft propulsion stages could also profitably employ such technology. Inert weight reductions are worth striving for, at least from the standpoint of MEO requirements. For instance, if inert weights appropriate to current vehicles were employed in the MEO calculations, the MEO requirements would increase by 15 to 40 percent depending on the mission-system characteristics.

Other structural items— Specific values of weights for the following items were obtained from the spacecraft design of reference 20. The design is based on 10-m-diameter modules. The following relationships are based on the point design modified to reflect a linear dependence with surface area, and they are considered independent of propellant loading and propellant type. These weights also are predicated on withstanding only the loads encountered after Earth orbit insertion.

Interstage structure: The following weight relationships apply to interstage structures that enclose nuclear engines, enclose chemical engines, and connect propellant tanks for which the upper tank has no engine, respectively:

$$W = 142(D_1 + D_2) \tag{15}$$

$$W = 65(D_1 + D_2) \tag{16}$$

$$W = 48(D_1 + D_2) \tag{17}$$

In these expressions and those that follow, W is in kilograms, and D_1 and D_2 are the diameters, in meters, of the interconnected modules.

Mission module adapter: This adapter connects the MM to the planet departure stage. The weight relationship is:

$$W = 30(D_1 + D_2) \tag{18}$$

Clustering structure: If two or more stages are connected in parallel, the weight of the required structure is:

$$W = 970(n - 1) \tag{19}$$

where n is the number of stages.

Docking mechanism: Weights for male and female docking mechanisms, including structure and actuating mechanisms, are estimated as:

$$\left.\begin{array}{ll} W = 47D & \text{(male)} \\ W = 130D & \text{(female)} \end{array}\right\} \tag{20}$$

Clustering structures and docking mechanisms have applicability, of course, only when rendezvous in Earth orbit is carried out. To avoid specification of particular Earth orbit operations in the MEO calculations, a single launch was assumed. Since the interstage weights depend on the module diameters, a value of 18 m was assumed for the Earth departure stage. A value of 10 m was assumed for the planet arrival and departure stages.

Planetary Aerobraker Systems

Although atmospheric braking to orbit about Mars and Venus is an attractive mode of capture when compared to propulsive braking, this mode results in a much more complex system. Moreover, the system requirements are very sensitive to the environment, vehicle characteristics, and trajectory parameters. Additional constraints are imposed by packaging, tolerable deceleration levels, and approach guidance accuracy.

As part of a detailed feasibility study (ref. 23), these rather complex interactions were analyzed in some detail and representative aerobraker performance capabilities determined. The primary results and applicable environmental and system characteristics are shown in figure 14, which depicts the equivalent aerobraker mass ratio (i.e., ratio of mass before atmospheric entry to mass after orbital capture) as a function of entry speed. The results thus include the propellant required to insert the spacecraft in the capture orbit after skipout. The entry speeds encompass the range appropriate to both Mars and Venus. Note that for a given speed the heat shield weight requirements are slightly less for Venus capture. While the peak heating rate is higher in the Venus atmosphere, the integrated heat flux is somewhat lower.

The aerobraker analysis included parametric variations in vehicle L/D, ballistic coefficient, atmospheric properties, and limiting criteria such as maximum deceleration or minimum pullup altitudes. Conservative estimates of the atmospheric properties were used on the assumption that adequate definition of the environment will be at hand before any actual designs are started: the analysis did not consider system design requirements covering the current wide range of uncertainties.

Midcourse Corrections and Attitude Control

Midcourse velocity corrections– From reference 24 the recommended total velocity requirement for each single leg of the mission (e.g., Earth-Mars and Mars-Venus) is 100 m/sec. Based on probable navigation accuracies and implementation errors, there is a virtual 100 percent probability that this requirement will not be exceeded. Earth-storable propellants were used in the MEO calculations for these maneuvers.

No firm requirement for maneuvers in planetary orbit has yet been defined, although for the Mars landing mission a 10-percent increase in the MEM ascent velocity requirements for rendezvous was included.

Because of the wide variation possible in Earth orbit operations (varying from a single launch of a large post-Saturn launch vehicle to multiple rendezvous), no Earth orbit velocity requirements are established. Thus, the MEO values calculated would apply to the spacecraft mass after any rendezvous maneuvers had been completed.

Attitude control requirements– Attitude control fuel requirements generally are very modest and essentially independent of the mass of propellant required at the planet. Rather, nearly all of the fuel is required to stabilize the mission module during the inbound leg, provided the control jets can be installed at the maximum distance from the spacecraft center of gravity. This is due, of course, to the relatively short moment arms which exist after the propellant tanks used at planet departure are jettisoned.

The spacecraft is considered nonrotating but with a centrifuge aboard for daily gravity simulation for the crew. Other disturbances considered are internal and external movement of the crew, solar pressure, and realinement of the spacecraft longitudinal axis toward the Sun each day. This is made necessary since it is likely that between alinements the attitude will be held to an inertial reference.

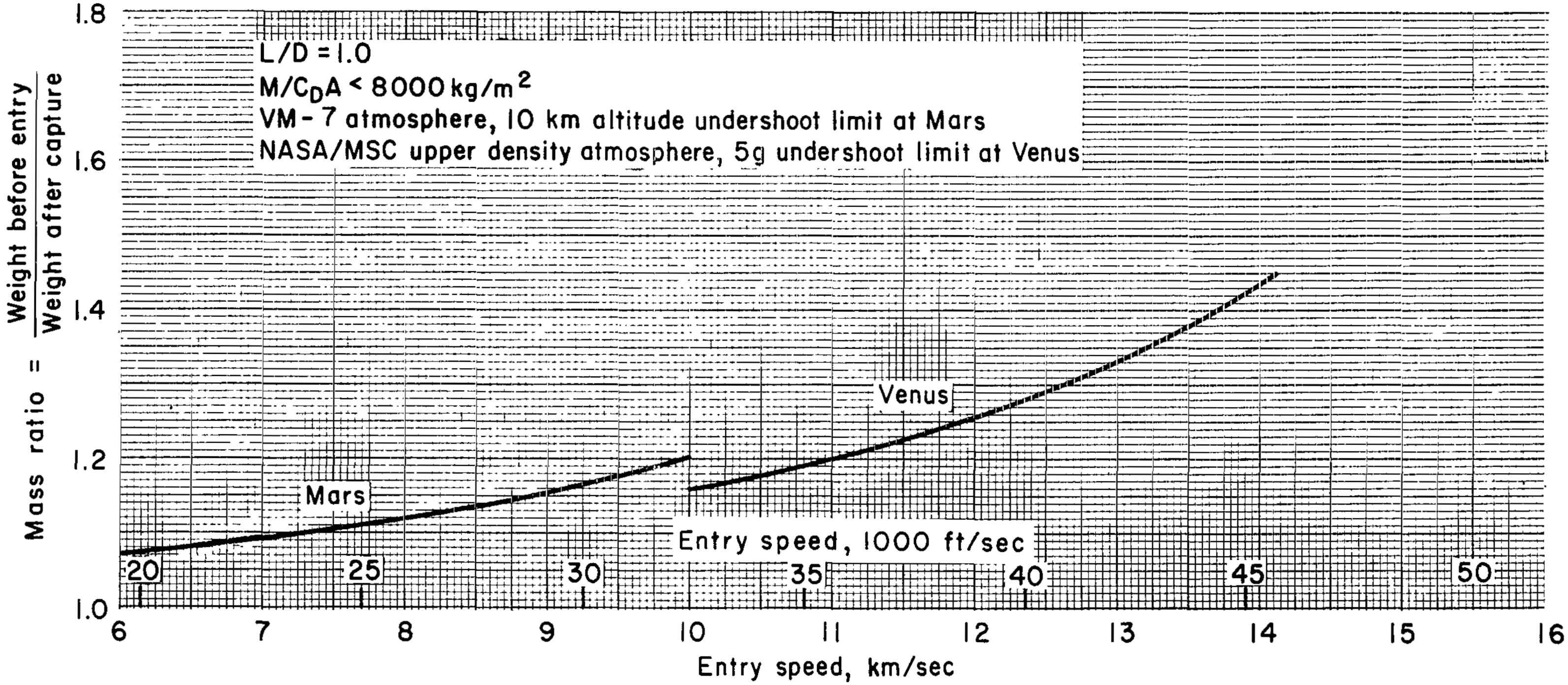

Figure 14.– Aerobraker effective mass ratio.

The alinement toward the Sun (or some other suitable reference) is assumed accurate within 1° in a limit cycle mode. During 1 hr each day, the accuracy increases to 0.1° for daily navigational sightings, photography of some event, etc.

The resulting fuel requirements are about 7.2 kg/day during the inbound leg and at least one order of magnitude less during the earlier phases of the mission. For convenience in the MEO calculations, an average value of 3.6 kg/day is taken as the requirement for the entire mission and added to the mission module expendables. The underlying assumptions in the determination of this requirement are considered reasonable; it should be pointed out, however, that the attitude control system characteristics may vary considerably as a function of vehicle and crew dynamics, experiment requirements, event schedules, and system configuration.

National Aeronautics and Space Administration
Moffett Field, Calif., 94035, Sept. 3, 1970

APPENDIX

SAMPLE CALCULATION OF REQUIRED MEO

A sample calculation of the required mass in Earth orbit is included here to provide a fuller understanding of the data presented in the body of the report and to aid those analysts who might wish to conduct analyses of a similar nature. The 1986 nominal Mars stopover lander mission using a Venus swingby is discussed. The trajectory data for this mission are given in table 1(b). With the use of the NACA propulsion system combination calculations for both nuclear and chemical stages, as well as those associated with atmospheric capture at Mars, can be illustrated.

The determination of the MEO begins with the total weight returned to Earth and progresses backward through the mission to obtain the gross weight in Earth orbit. The calculation is therefore divided into several parts which correspond to maneuvers or legs of the mission.

TOTAL WEIGHT PRIOR TO SEPARATION OF MISSION MODULE (MM) AND EARTH REENTRY MODULE (ERM)

From equation 11 and figure 11,

$$\begin{aligned} MM &= 24170 + (12.2)*(520 \text{ days}) \\ &= 30{,}514 \text{ kg} \end{aligned} \tag{A1}$$

$$ERM = 5750 \text{ kg} \tag{A2}$$

TOTAL WEIGHT JUST AFTER DEPARTURE FROM MARS (WDM)

To obtain WDM, expandable supplies at a rate of 16.6 kg/day must be added to the MM and ERM weight. Also, midcourse velocity correction requirements must be considered.

Expendables:

$$\begin{aligned} EXP &= (16.6 \text{ kg/day})*(170 \text{ days}) \\ &= 2822 \text{ kg} \end{aligned} \tag{A3}$$

Midcourse correction requirements:

Because of the small weight requirements for the midcourse correction, a simplified calculation is used to determine the weight requirements. The values obtained are conservative, however.

The ΔV required is 100 m/sec. The propulsion system selected is N_2O_4/A50 which has an I_{sp} = 330 sec. Therefore

$$\mu = \exp \frac{100 \text{ m/sec}}{g(330 \text{ sec})} = \exp \frac{0.1}{0.0098\ (330)} = 1.0314 \tag{A4}$$

The engine weight is fixed at 45 kg (100 lb). Then

$$\mu = \frac{W_{propellant} + MM + ERM + EXP + W_{engine}}{MM + ERM + EXP + W_{engine}} \tag{A5}$$

where $W_{propellant}$ = 1229 kg. Then from equation (12)

$$W_{tank} = \frac{0.21(1255)^{0.9}}{(1.33)^{0.533}} = \frac{(0.21)(603.0)}{1.642} = 109 \text{ kg} \tag{A6}$$

Adding W_{tank} (109), W_{engine} (45), and $W_{propellant}$ (1229), the total weight for the midcourse correction is 1383 kg. Then,

Midcourse	1,383 kg
EXP	2,822
MM	30,514
ERM	5,750
WDM	40,469 kg (89,234 lb)

TOTAL WEIGHT PRIOR TO DEPARTURE FROM MARS ORBIT (WDMO)

The propulsion system selected for the Mars departure maneuver is the cryogenic (O_2/H_2) system with the characteristics given in table 6. The stage thrust-to-weight (T/W) ratio is optimized for this maneuver. That is, a computer program assesses the effects of gravity losses and determines the T/W that gives the minimum gross weight for the stage. The optimization procedure for the T/W is not discussed since the primary purpose is to illustrate the use of the scaling laws. The basic procedure relies on empirical equations of the gravity loss data contained in references 25 and 26. The optimum T/W is 0.38 for this Mars departure maneuver.

The relevant equations required to obtain the stage weights are given below. Note that these are transcendental equations and hence cannot be solved algebraically. The solution found by computer iteration is given.

$$\mu = \exp \frac{(\Delta V + V_G)}{(G_o)*(I_{sp})} \tag{A7}$$

where

ΔV velocity requirement from table 1(b)

V_G gravity loss

G_o acceleration of gravity of 0.0098 km/sec^2

I_{sp} propellant specific impulse

$$\mu = \frac{\text{WDM + inert weight + ignition propellant}}{\text{WDM + inert weight}} \tag{A8}$$

where

Inert weight = tank weight + engine weight + insulation weight

Note that the meteoroid shield weight is not included in the inert weight equation since it is assumed to be jettisoned just prior to ignition of the stage.

Tank weight:

One tank, as opposed to multiple tanks, is used in this example. As noted above, the solution to the relevant equations cannot be determined algebraically. The solution in terms of total propellant loading is therefore given without illustrating the iteration technique. The total propellant weight is 27,825 kg (61,352 lb); it includes both ignition and boiloff propellant and hence represents the full propellant weight at Earth departure. Storage of this amount of propellant requires a small cylindrical tank with hemispherical ends with a radius of 2.06 m (6.75 ft) and a length of 4.12 m (13.5 ft). The surface area of the side walls and one hemispherical end is therefore 79.79 m^2 and the forward bulkhead area next to the MM is 26.6 m^2.

From equation (14a)

$$W_{tank} = \frac{0.08(27{,}825)^{0.9}}{(0.32)^{0.533}} + K \text{ kg} = 2123 \text{ kg} \tag{A9}$$

where K = 500 + 154 kg for interstage structure.

Engine weight:

The optimum thrust is 26,821 kg (59,140 lb) and therefore from table 6, the engine weight is 268 kg.

Meteoroid shield weight:

From equation (9) and table 3,

$$\bar{t}_{act} = 10 \left[\frac{(79.79)(380)}{(400)(250)}\right]^{1/3} = 6.72 \frac{kg}{m^2} \tag{A10}$$

Thus the meteoroid shield weight = (6.72)*(79.79)

= 536 kg (1182 lb)

Tank insulatior weight:

From equation (1) and table 2, the insulation requirement for the side walls and forward bulkhead (which is next to the mission module for the Mars departure stage) is given by the following equations:

Sidewall ($A = 79.79\ m^2$):

$$W_{ins} = A\sqrt{\frac{\rho k(1 + 1)(T_s)}{(108\ kcal/kg)(1.614)(1.19)(1.611)}} \tag{A11}$$

where

$$\rho = 48\ kg/m^3$$

$$k = 9.26\times10^{-4}\ kcal/(day\text{-}m\text{-}^\circ K)$$

$$T_s = (30)(195\text{-}20) + (320)(117\text{-}20) + (30)(220\text{-}20) = 42290\ (days - {}^\circ K)$$

Forward bulkhead ($A = 26.60\ m^2$):

$$W_{ins} = A\sqrt{\frac{\rho k(1 + 1)(T_F)}{(108\ \text{kcal/kg})(1.614)(1.19)(1.611)}} \tag{A12}$$

where

$$T_F = (294\text{-}30)(30 + 320 + 30) = 100{,}320(\text{days-}^\circ\text{K})$$

$$\text{total insulation} = 405\ \text{kg}\ (893\ \text{lb}) \tag{A13}$$

Note that no gravity losses are included in the mass ratios. Since the losses have not yet been determined, the assumption is made that the effect on the insulation weight is small and hence can be neglected.

Boiloff propellant:

It should be made clear that the inert weights are based on the total propellant loading at Earth departure. Some propellant, however, will have boiled off prior to ignition. From equations (2) and (A13) the weight of this boiloff is:

$$\begin{aligned} W_{bo} &= (405\ \text{kg})*(1.614)(1.19)(1.611) \\ &= 1253\ \text{kg}\ (2763\ \text{lb}) \end{aligned} \tag{A14}$$

This boiloff is assumed to occur in three phases: Earth orbit, outbound leg, and Mars orbit. The total boiloff for the Mars departure stage is therefore divided into three parts according to a time-temperature ratio as follows:

Mars orbit $\quad \dfrac{(30)(195\text{-}20)}{T_S} = 0.12$

Outbound leg $\quad \dfrac{(320)(117\text{-}20)}{T_S} = 0.74$

Earth orbit $\quad \dfrac{(30)(220\text{-}20)}{T_S} = 0.14$

The gross weight of the spacecraft immediately prior to ignition of the Mars departure stage (WDMO) is then given as follows:

WDM	40,469 kg
Mars departure ignition propellant	26,572
Mars departure tank weight	2,123
Mars departure engine weight	268
Mars departure meteoroid shield	536
Mars departure insulation weight	405
WDMO	70,373 kg (155,172 lb)

The gravity loss for the Mars departure stage is 0.01 km/sec so that $\mu = 1.614$ from equation (A7). This value agrees with the value of μ from equation (A8) so that the iterative solution is valid.

TOTAL WEIGHT AFTER MARS ARRIVAL (WAMA)

The boiloff weight for the Mars departure stage must be included here. The total weight off-loaded in Mars orbit from figure 12, based on the use of cryogenic propellants and a circular orbit, is:

MEM	32,500 kg
Probes	9,000
Rover	11,000
Housing	3,460
PEM	55,960 kg

The expendables consumed during the 30-day stay time at Mars are (16.6)*(30) days = 498 kg.

WAMA is therefore:

WDMO	70,373 kg
Mars departure stage boiloff	150
PEM	55,960
Expendables	498
WAMA	126,981 kg (279,993 lb)

TOTAL WEIGHT BEFORE MARS CAPTURE (WBMC)

Aerodynamic capture is used at Mars. From figure 14, the aerobraker effective mass ratio for an entry speed of 9.6 km/sec is 1.19; therefore, the WBMC is (1.19)*(126,981) = 151,107 kg (333,192 lb).

TOTAL WEIGHT AFTER EARTH DEPARTURE (WAED)

As before, the following weights are added during the outbound leg.

WBMC	151,107 kg	
Venus probes	2,500	(fig. 12)
Expendables	5,312	(16.6 * 320)
Mars departure stage propellant boiloff	927	
Midcourse correction	10,221	
WAED	170,067 kg	

The midcourse correction is 200 m/sec since the outbound leg of the mission is composed of two trajectories, Earth-Venus and Venus-Mars. The calculation of the weight requirement is exactly the same as for the inbound leg.

TOTAL WEIGHT IN EARTH ORBIT (MEO)

The propulsion system used for Earth escape is a nuclear stage with the characteristics given in table 6. The approach taken is to optimize the thrust to weight of the stage by selecting the appropriate number of nuclear engines. Since this procedure requires iteration, only the solution is given here. Two engines represent the optimum number, resulting in a total engine weight of 21,000 kg and a thrust of 68,000 kg. The total propellant loading is 145,450 kg of which 2,725 kg is boiloff propellant. A 73-ft-long cylindrical tank with hemispherical bulkheads with a 16.5-ft radius (33.0 diam) contains this volume of propellant. The total tank surface area is 1,017 m^2.

Tank weight: From equation (14a) where A = 0.08 and K = 500 plus 724 kg for interstage structure,

$$W_{tank} = \frac{0.08(145{,}450)^{0.9}}{(0.068)^{0.533}} + 1224 \text{ kg} = 16{,}077 \text{ kg}$$

Meteoroid shield:

$$W_{shield} = A \ 6.0 \text{ kg/m}^2 \left[\frac{A \ 30}{(400 \ m) 30}\right]^{1/3} = 8372 \text{ kg}$$

Insulation:

$$W_{ins} = A\sqrt{\frac{\rho k(1 + 1)(T_S)}{(108 \text{ kcal/kg})(1.611)}} = 1790 \text{ kg}$$

where ρ and k are as before and

$$T_S = (30) * (220\text{-}22) = 5940 \text{ (days - °K)}$$

Boiloff propellant:

$$W_{bo} = 1790(1.611)\text{kg} = 2880 \text{ kg}$$

All this boiloff occurs during the 30 days in Earth orbit.

The gross weight of the spacecraft immediately prior to ignition of the Earth escape stage (WBED) is given as follows:

WAED	170,067 kg
Earth departure ignition propellant	142,725
Earth departure tank weight	16,077
Earth departure engine weight	21,000
Earth departure meteoroid shield weight	8,372
Earth departure insulation weight	1,790
WBED	360,031 kg

The gravity loss for the Earth escape stage is 0.37 km/sec so that $\mu = 1.68$.

The following weights must be added to WBED in order to account for the 30-day time period in Earth orbit.

WBED	360,031 kg
Mars departure stage propellant boiloff	176
Earth escape stage propellant boiloff	2,880
Expendables	498
MEO	363,586 kg

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